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ADDENDUM TWO

Study on the Feasibility of V/STOL Concepts for Short Haul Transport Aircraft

Prepared Under CONTRACT NO. NAS 2-3035

For

NASA AMES RESEARCH CENTER
MOFFETT FIELD, CALIFORNIA

LOCKHEED-CALIFORNIA COMPANY • BURBANK, CALIFORNIA
A DIVISION OF LOCKHEED AIRCRAFT CORPORATION

STUDY ON THE FEASIBILITY OF V/STOL CONCEPTS FOR SHORT HAUL TRANSPORT AIRCRAFT

ADDENDUM REPORT LR-20573

MARCH 1967

Prepared Under Contract No. NAS 2-3035 by
THE LOCKHEED-CALIFORNIA COMPANY

for

NASA AMES RESEARCH CENTER
MOFFETT FIELD, CALIFORNIA

N 67 32478

FOREWORD

This document consists of addenda to Lockheed Report 19586, "Study on the Feasibility of V/STOL Concepts for Short Haul Transport Aircraft - Research Report". This is a second addendum report to LR 19586 and describes additional short haul transport studies made at the Lockheed-California Company between 30 June 1966 and 1 March 1967 as an extension of Contract NAS 2-3035 with the NASA Ames Research Center. This work was concerned with standardized weight estimates and noise sensitivity analyses as well as additional development studies of the tilt and stopped rotor concepts.

The first addendum report, LR 19585, dated 30 March 1966, consisting of three volumes, reflects an earlier phase of development of the tilt and stopped rotor concepts and includes addenda for the other concepts studied.

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1. WEIGHT STANDARDIZATION

During the Lockheed study, component weights were estimated from statistical data and from studies of unique structural aspects of each concept. The Naval Air Systems Command Weight Control Branch evaluated the weights of the 60-passenger VTOL Tilt Rotor, 1000 ft STOL Fan-In-Wing, 2000 ft STOL Deflected Slipstream, 2000 ft STOL Jet Flap, and VTOL Lift/Cruise Fan. Component weight estimates were derived based on their methods. The NASC weight estimates differed from Lockheed's in some areas, and the weight estimates were standardized based on the NASC estimates. After the weights were standardized, direct operating costs for various stage lengths were calculated. These DOC's were then compared with the DOC's previously derived to describe a band of DOC versus stage length for the described weight sensitivity.

Figure 1-1 shows Lockheed's and NASC's weight estimates along with the revised NASC estimates for the five concepts evaluated. The initial NASC estimates were derived for the same gross weight as the Lockheed estimates and the difference in these estimates is indicated in the fuel weight available as shown in Figure 1-1.

During the follow-on study, the Tilt Rotor configuration was re-evaluated and revised. A weight breakdown for the revised configuration is shown in Figure 1-2 along with NASC weight estimates and revised gross weight. In order to standardize the weight estimates, Lockheed and NASC personnel conferred to resolve differences in computational methods. Some of these differences were resolved and agreement was reached on how to determine component weight variations with gross weight; this was required to determine the revised gross weights based on the NASC estimates.

When the five concepts were scaled up in gross weight the wing loading, thrust/weight, tail area/weight, and disc loading were held constant so that vehicle performance did not change with respect to cruise altitude and cruise speed for the relatively small weight changes. The fuel required for the 500-mile stage length was determined from fuel-required versus gross-weight curves developed with the above parameters held constant. The revised fuselage

WE

	VTOL TILT ROTOR		
	LOCKHEED	NASC	REVISE NASC
GROSS WING AREA (FT^2)	690	690	760
HORIZ. TAIL AREA (FT^2)	264	264	291
VERT. TAIL AREA (FT^2)	135	135	149
THRUST OR HORSEPOWER/ENGINE	3090	3090	3405
PROPELLER OR ROTOR DIAMETER (FT)	49.93	49.93	52.41
LIFT FAN - TIP TURBINE DIAM (IN.)	-	-	-
CRUISE FAN - FAN DIAMETER (IN.)	-	-	-
WING	4330	5245	5780
TAIL	1530	1460	1630
FUSELAGE	6310	7530	7030
LANDING GEAR	2300	2435	2650
NACELLES	2170	2020	2180
CONTROLS AND HYDRAULICS	2630	3140	3350
ENGINES AND CRUISE FANS	2240	2240	2470
AIR INDUCTION SYSTEM		160	165
EXHAUST SYSTEM		120	125
LUBE SYSTEM		180	180
ENGINE ACCESSORIES	1120		
FUEL SYSTEM	450	450	450
ENGINE CONTROLS		130	130
STARTING SYSTEM		160	160
PROPS, ROTORS, OR FANS	4845	5340	5880
HOT GAS SYSTEM	-	-	-
MAIN GEARBOXES	3720		
CROSS SHAFT GEARBOXES	310	4610	4840
CLUTCHES	170		
CROSS SHAFTING	295		
INSTRUMENTS	410	530	530
ELECTRICAL & ELECTRONICS	1800	2440	2440
FURNISHINGS	5010	4900	4900
AIR COND. & ANTI-ICING	1625	1520	1520
AUXILIARY POWER UNIT	360	500	500
AUXILIARY GEAR		40	40
WEIGHT EMPTY	41,625	45,150	46,950
CREW	520	520	520
MISC. USEFUL LOAD	260	700	700
ENGINE OIL	150	190	190
UNUSABLE FUEL	75	40	50
PAYLOAD	13,200	13,200	13,200
ZERO FUEL WEIGHT	55,830	59,800	61,610
USABLE FUEL AVAILABLE	4790	1000	5350
GROSS WEIGHT	60,800	60,800	67,000

FIGURE 1-1

STANDARIZATION - WEIGHT STATEMENTS

1000-FT FAN IN WING			2000-FT DEFL. SLIPSTREAM			2000-FT JET FLAP			VTOL LIFT/CRUISE FAN		
OCKHEED	NASC	REVISED NASC	LOCKHEED	NASC	REVISED NASC	LOCKHEED	NASC	REVISED NASC	LOCKHEED	NASC	REVISED NASC
1069	1069	1155	832	832	896	843	843	898	798	798	866
330	330	357	237	237	255	125	125	133	199	199	216
293	293	317	211	211	227	172	172	183	153	153	166
6488	6488	7010	1275	1275	1370	6800	6800	7240	7300	7300	7920
-	-	-	14.07	14.07	14.6	-	-	-	-	-	-
60.0	60.0	62.4	-	-	-	-	-	-	94.2	94.2	98.1
-	-	-	-	-	-	-	-	-	64.8	64.8	67.45
6695	8020	8670	4630	4420	4760	7520	7390	7870	4290	4205	4560
2780	3050	7320	1320	1370	1490	1510	1670	1790	1600	1560	1710
6865	7950	7300	5310	7360	6710	6700	7950	8050	6970	7650	7750
2580	2690	2880	1750	2080	2220	2400	2550	2695	2740	2950	3175
1620	1170	1240	1085	1300	1380	2110	2040	2145	4940	5000	5335
1960	2410	2535	1680	1830	1920	1950	1810	1885	2020	2450	2585
2990	3460	3740	1120	940	1010	4490	4440	4730	6845	7820	7900
	100	105		60	65		140	145		810	840
	250	260		140	145		180	185		70	75
	140	140		180	180		140	140		210	210
360	480	480	560			540			1685	630	650
540	560	560	430	410	410	500	500	500	515	515	515
140	120	120		120	120		120	120		180	180
	160	160		160	160		140	140		210	210
2030	1600	1730	2100	1240	1335	-	-	-	3040	3700	4015
1830	830	860	-	-	-	2130	2420	2495	1915	1900	1615
-	-	-	530			-	-	-	-	-	-
-	-	-	220	1520	1575	-	-	-	-	-	-
-	-	-	145			-	-	-	-	-	-
-	-	-	195			-	-	-	-	-	-
430	500	500	420	490	490	430	460	460	550	530	530
1800	2340	2340	1800	2140	2140	1800	2360	2360	1800	2490	2490
5060	4950	4950	4925	4430	4430	5070	4580	4580	5070	4790	4790
1575	1350	1350	1025	1400	1400	1575	1440	1440	1570	1390	1390
370	500	500	345	500	500	360	500	500	375	500	500
	40	40		40	40		40	40		40	40
9,625	42,670	43,780	29,590	32,130	32,480	39,085	40,870	42,270	45,925	49,060	51,065
520	520	520	520	520	520	520	520	520	520	520	520
160	700	700	260	700	700	260	700	700	260	700	700
105	70	70	60	130	130	110	130	130	175	80	80
210	200	200	50	40	40	150	70	75	175	90	90
3,200	13,200	13,200	13,200	13,200	13,200	13,200	13,200	13,200	13,200	13,200	13,200
3,920	57,360	58,470	43,680	46,720	47,070	53,325	55,490	56,895	60,255	63,650	65,655
3,980	10,540	14,930	3220	180	3430	9875	7710	10,405	11,545	8150	12,245
'7,900	67,900	73,400	46,900	46,900	50,500	63,200	63,200	67,300	71,800	71,800	77,900

FIGURE 1-2

WEIGHT STANDARIZATION - REVISED TILT ROTOR
(pounds)

	LOCKHEED ESTIMATE	NASC ESTIMATE	REVISED NASC
WING	4,755	5,760	6,300
TAIL	1,275	1,450	1,610
FUSELAGE	6,320	7,030	7,030
LANDING GEAR	2,390	2,500	2,705
NACELLES	2,600	2,450	2,630
CONTROLS & HYDRAULICS	2,985	3,300	3,400
ENGINES	2,680	2,680	2,930
AIR INDUCTION SYSTEM		165	170
EXHAUST SYSTEM		125	130
LUBE SYSTEM		180	180
ENGINE ACCESSORIES	820		
FUEL SYSTEM	465	465	465
ENGINE CONTROLS		130	130
STARTING SYSTEM		160	160
ROTORS	5,520	5,700	6,240
MAIN GEARBOXES	4,280		
CROSS SHAFT GEARBOXES	370	5,350	5,600
CLUTCHES	220		
CROSS SHAFTING	380		
INSTRUMENTS	410	530	530
ELECTRICAL & ELECTRONICS	1,800	2,440	2,440
FURNISHINGS	5,040	4,900	4,900
AIR CONDITIONING & ANTI-ICING	1,660	1,520	1,520
AUXILIARY POWER UNIT	365	500	500
AUXILIARY GEAR		40	40
WEIGHT EMPTY	44,335	47,375	49,610
CREW	520	520	520
MISC. USEFUL LOAD	260	700	700
ENGINE OIL	185	190	190
UNUSABLE FUEL	100	60	70
PAYLOAD	13,200	13,200	13,200
ZERO FUEL WEIGHT	58,600	62,045	64,290
USABLE FUEL AVAILABLE	6,400	2,955	6,910
GROSS WEIGHT	65,000	65,000	71,200

weights agreed on with NASC were held constant as gross weight was increased. These revised weights are based on the wing being lowered from 15 to 18 inches on the configuration studied. These configurations then provided a 72-inch head clearance at the wing spars.

The original gross weights, along with the revised gross weights and percent changes are summarized in Figure 1-3. The major weight differences remaining after negotiation with NASC are shown in Figure 1-4. The paragraphs following discuss these differences.

The basic difference between the weight estimates on the Tilt Rotor wing is that NASC penalizes a conventional wing 54%, or 3% of the gross weight, and Lockheed penalizes the wing 35%, or 1.9% of the gross weight, for the VTOL capability.

On the Fan-In-Wing wing weight the difference is in the penalty associated with mounting the fans in the wing. NASC penalizes a conventional wing 42%, or 3.5% of the gross weight, while Lockheed's penalty (after making a structural study) is 18.5% or 1.54% of the gross weight. The wing weights for the other configurations are in good agreement.

The body weight estimates differ due to differences in the methods used. NASC estimates the basic fuselage shell statistically and adds penalties for pressurization, windows, landing gear, doors, etc. by comparison to contemporary aircraft. Lockheed estimates the complete fuselage statistically and adds penalties for design features that deviate from the contemporary aircraft used for the statistical method. Lockheed feels that this method is better for this type study. When a large portion of the weight consists of judgment penalties, the effect of design parameter variation is lost since these penalties tend to remain relatively constant for fuselages of the same size. The end result is that the body weights do not vary significantly from one configuration to another using the NASC method and do vary significantly using Lockheed's method since there are wide variations in cruise speed, cabin pressurization, and gross weight among the configurations studied.

The difference in the control system weight estimates is in the increments added for STOL and VTOL capability. At this stage of the design the control systems are rather nebulous and must be predicted by Phase I statistical methods. A conventional flight control system is estimated in this manner, and increments are added for the STOL or VTOL capability.

Figure 1-3

WEIGHT COMPARISONS

60 Passenger - 500 Statute Mile Range

Takeoff Length	Configuration	Original Gross Weight (pounds)	Revised Gross Weight (pounds)	Percent Change
VTOL	Tilt Rotor	65,000	71,200	+9.5
1000 ft	Fan In Wing	67,900	73,400	+8.1
2000 ft	Deflected Slipstream	46,900	50,500	+7.7
2000 ft	Jet Flap	63,200	67,300	+6.5
VTOL	Lift/Cruise Fan	71,800	77,900	+8.5

Figure 1-4

MAJOR WEIGHT DIFFERENCES AFTER NEGOTIATION

WITH NASC RELATIVE TO INITIAL COMPONENT WEIGHTS

60 Passenger - 500 Statute Mile Range

(Weight difference in pounds)

Configuration	Wing	Body	Controls	Propul-sion	Electrical & Electronics	Useful Load
Tilt Rotor	+1005	+710	+315	----	+640	+440
Fan In Wing	+1325	+435	+450	----	+540	+440
Deflected Slipstream	----	+1400	+150	----	+340	+440
Jet Flap	----	+1350	-140	+420	+560	+440
Lift/Cruise Fan	----	+780	+430	+1505	+690	+440

The propulsion system weights are in good agreement except for the Lift/Cruise Fan configuration; 1091 pounds of the 1505 pounds consists of disagreement in gas generator and fan weights. Lockheed used General Electric data for scaling engine and fan weights, while NASC used a consolidated method consisting of constant thrust/weight ratio for engines and a similar system for lift and cruise fans. The primary difference is in the estimates of cruise fan weights.

The electrical, electronic, and useful load weights are more a subject of design philosophy than weight analysis. An electronic list was derived and is the basis for Lockheed's estimates. The useful load question consists of whether food service should be required or whether beverage service only is sufficient; also, whether pillows and magazines should be provided for the passengers on this type of aircraft. It is interesting to note that the electrical, electronic, and useful load items account for approximately 3% of the gross weight difference between Lockheed's and NASC's estimates after a growth factor is applied.

Figure 1-5 summarizes the direct operating cost comparisons for the original and revised weights for the 500-mile stage lengths. Figure 1-6 tabulates the direct operating cost for various stage lengths for the given weight sensitivity. The parametric designs are also shown so that a wider range of weights may be evaluated and their effect on DOC examined.

FIGURE 1-5
DIRECT OPERATING COST - CENTS/AVAILABLE SEAT MILE
(500 ST. MILE STAGE LENGTH)

CONFIGURATION	ORIGINAL WEIGHTS	REVISED WEIGHTS	PERCENT CHANGE
TILT ROTOR	2.67	2.83	+ 6.0
FAN IN WING	2.67	2.81	+ 5.2
DEFLECTED SLIPSTREAM	1.96	2.11	+ 7.7
JET FLAP	2.26	2.36	+ 4.4
LIFT/CRUISE FAN	2.87	3.03	+ 5.6

FIGURE 1-6
DIRECT OPERATING COST VS STAGE LENGTH

300 PRODUCTION UNITS - 2000 HOURS UTILIZATION - 60 AVAILABLE SEATS
DIRECT OPERATING COSTS SHOWN IN CENTS PER AVAILABLE SEAT MILE

GROSS WT.		STAGE LENGTH (MILES)				
		25	50	100	200	500
	TILT ROTOR					
58,200	PARAMETRIC DESIGN	9.42	5.71	3.76	2.79	2.27
65,000	FINAL DESIGN	11.08	6.71	4.42	3.28	2.67
71,200	NASC WEIGHTS	11.76	7.13	4.69	3.48	2.83
	LIFT/CRUISE FAN					
70,000	PARAMETRIC DESIGN	13.82	8.08	5.16	3.64	2.82
71,800	FINAL DESIGN	14.06	8.22	5.25	3.70	2.87
77,900	NASC WEIGHTS	14.74	8.63	5.51	3.90	3.03
	DEFLECTED SLIPSTREAM					
45,600	PARAMETRIC DESIGN	6.84	4.29	2.96	2.30	1.92
46,900	FINAL DESIGN	6.98	4.38	3.02	2.35	1.96
50,500	NASC WEIGHTS	7.48	4.72	3.24	2.53	2.11
	JET FLAP					
59,500	PARAMETRIC DESIGN	10.56	6.32	4.06	2.92	2.18
63,200	FINAL DESIGN	10.92	6.54	4.21	3.03	2.26
67,300	NASC WEIGHTS	11.28	6.79	4.38	3.17	2.36
	FAN IN WING					
63,700	PARAMETRIC DESIGN	12.42	7.42	4.84	3.44	2.54
67,900	FINAL DESIGN	13.05	7.80	5.09	3.62	2.67
73,400	NASC WEIGHTS	13.67	8.17	5.34	3.80	2.81

2. TILT ROTOR AND STOPPED ROTOR OPTIMIZATION

A more refined optimization study has been performed on the tilt rotor configuration and four stopped rotor configurations incorporating what is felt to be more realistic propeller and rotor characteristics. Configurations considered for the stopped rotor vehicles were a single rotor, stopped, folded, and stowed; and a twin rotor stopped, folded, and trailed. Each of these rotor systems was evaluated with the use of both propellers and jet propulsion for cruise flight.

The parametric values of the variables considered in the study are listed in Figure 2-1. For the twin rotor configurations the wing span was fixed by the required rotor radii and necessary clearances. This span in turn determines wing area and therefore wing loading at any given gross weight.

The required engine sizes for the various parametric vehicles were determined using the figure of merit values of Figure 2-2. These values of figure of merit are considered representative of current rotor technology. It is felt, however, that a serious development program applying some of the principles of propeller design to rotor design could significantly raise the values of the figure of merit at higher tip speeds. An additional study is presented later in this report showing the effect on aircraft characteristics of the projected increase in figure of merit by this application of propeller technology. It is emphasized, however, that the parametric study was based on the curve of Figure 2-2.

To understand the significance of the figure of merit level used in the stopped rotor and tilt rotor parametric studies it is necessary to clearly define the ground rules assumed for the parametric studies and the limitations of the rotor and propeller analyses employed to establish this level. It is also necessary to state clearly the significance of the figure of merit itself in its relation to rotary wing hover performance.

Figure of merit is defined as the ratio of rotor ideal induced power to the total power required in hover (profile power + induced power). The power required by a rotor in hover is primarily determined by the disc loading which effects the available thrust per horsepower as shown in the following equation:

FIGURE 2-1
PARAMETRIC VALUES USED FOR STOPPED AND TILT ROTOR OPTIMIZATION

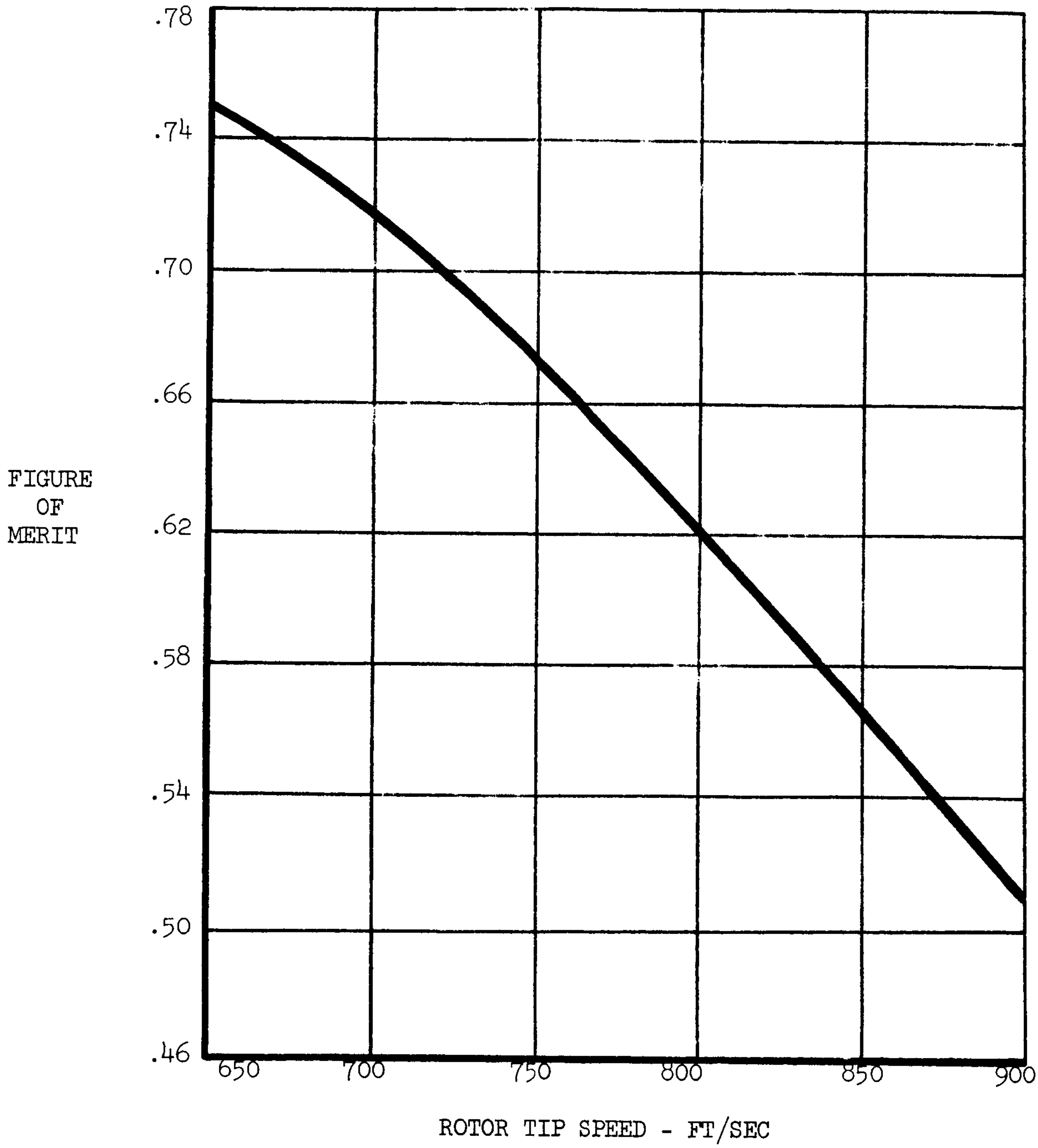
A/C CONCEPT	GROSS WEIGHT (1000 LBS)	ASPECT RATIO	WING LOADING W/S (LB/FT^2)	DISK LOADING (LB/FT^2)	PROP DIAMETER (FT)	ROTOR TIP SPEED (FT/SEC)	THRUST TO WEIGHT T/W	YAW ACCELL REQUIREMENT $\ddot{\phi}$
SINGLE STOWED-ROTOR PROP	65,75,85	6	80,100,120	7,11,15	14,16,18	700 800 900	(1)	(1)
SINGLE STOWED-ROTOR JET	65,75,85	6	80,100,120	7,11,15		700 800 900	(1)	(1)
TWIN TRAILED ROTOR PROP	65,75,85	4,6,8	-----	7,11,15	14,16,18	700 800 900	(1)	(1)
TWIN TRAILED ROTOR JET	65,75,85	4,6,8	-----	7,11,15	-----	700 800 900	(1)	(1)
TILT ROTOR	65,75,85	6	-----	-----	46,56,66	700 800 900	(1)	(1)

(1) DEFINED BY CONTRACT REQUIREMENTS

FIGURE 2-2

FIGURE OF MERIT VS. ROTOR TIP SPEED

$C_T/\sigma = .1$



$$\frac{T}{SHP} = 550\, M \left(\frac{2\rho}{DL} \right)^{\frac{1}{2}}$$

Where:

T = thrust required for hover

SHP = shaft horsepower required for hover

M = figure of merit

ρ = air density

DL = disc loading

At a constant figure of merit, as the disc loading increases, the engine size requirement increases. This is a first order effect on hover thrust per horsepower. The effect of disc loading on figure of merit is a second order effect. Since the figure of merit is the ratio of the rotor ideal induced power to total power, and the profile power at a given tip speed is constant for a given blade loading $C_{T/\sigma}$; the figure of merit increases with disc loading as the induced velocity increases. This explains why the XC-142 propeller with a disc loading of 48.3 lb/ft^2 has a figure of merit of .79 and produces 4.31 pounds thrust/SHP, while a rotor with a disc loading of 13 lb/ft^2 and a figure of merit of .69 produces 7.27 pounds thrust/SHP.

Two analyses were used to establish the figure of merit level for the stopped rotor and tilt rotor parametric studies. There are no significant differences in theory between these two methods, namely, the Lockheed hover analysis and the Hamilton Standard propeller analysis. The primary differences between these analyses are in the two-dimensional airfoil section data range currently available in each, and in the geometries and operating conditions for which these data were synthesized.

The Hamilton Standard propeller analysis contains data for NACA 16-series and 64-series airfoil sections. Primary propeller analysis is normally carried out with the 16-series airfoil data. These data have been normalized to produce a smooth family of curves which represent incompressible performance for a full family of thickness ratios from .00 to .36. Correction for camber is introduced from normalized curves for design lift coefficients to 1.0. Compressibility effects on C_l and C_d is accounted for through a Von Karman correction up to a critical Mach number which is obtained from normalized curves as a function of thickness ratio, design C_L, chord to diameter ratio, and radial station. Above this critical Mach number an emperical compressibility factor is applied to a C_d/C_l ratio. All of these data have been synthesized to reflect three-dimensional effects from results of tests of existing propellers, which means that computations for non-standard propeller geometries and operating conditions might yield somewhat erronious results.

The Lockheed hover analysis contains data for NACA 00-Series airfoils for thickness ratios from .06 to .12. At the time that the study under discussion was performed, there was no capability to reflect the effects of camber. Aside from the difference in basic airfoil section family, the primary difference between these data and the data in the propeller analysis is in the way in which compressibility is accounted for. The 00-series data is a direct function of Mach number and angle of attack synthesized from rotor test data rather than propeller test data.

Considering the limitations listed above, it was decided to use the rotor analysis for the parametric study since it was felt that this analysis would provide the most accurate state-of-the-art results for hovering flight. However, upon re-examining the stopped rotor and tilt rotor vehicle requirements and considering the poor high speed characteristics of the uncambered 00-series airfoils which effect performance at high tip speeds, it became evident that by tailoring the airfoil sections by addition of camber as well as further optimization of blade twist, thickness distribution, and planform taper beyond the capability of the hover analysis with its current data format, a substantial increase in hover performance could be obtained at high tip speeds. Realizing that some loss in accuracy might result, the propeller analysis was used to establish reasonable performance levels for these rotors to take advantage of the flexibility of the normalized airfoil data available in this method. This investigation showed that a figure of merit of .69 for a tip speed of 900 feet per second is indeed reasonable for the moderate disc loadings at which these vehicles will be operating. This conclusion is borne out by comparison of data in Reference 10 which clearly shows the gains in figure of merit which can be made at high tip speeds by proper airfoil selection, especially when Mach number effects can be delayed. The effect on both the tilt rotor and stopped rotor configurations of application of these higher figure of merit values is shown later in this report.

Figure 2-3 shows a comparison of the best stopped rotor vehicle from the current study and the best configuration of the study previously reported. Due to the lower rotor figure of merit (.621 instead of .67) the gross weight of the vehicle increases to 78,200 pounds.

Figure 2-4 shows a typical set of results of the parametric study for one rotor tip speed and wing loading. It may be seen from the lower curves that the minimum DOC airplane has a rotor disc loading of 7 and a propeller diameter of 16 feet. The propeller used in the study had an activity factor of 200; however, later examination indicated some cruise benefits to be gained from a lower activity factor. An additional study was performed which led to the selection of a propeller with an activity factor of 140. The resulting airplane is indicated by the square points on the curves.

The effect of varying wing loading on various parameters is shown in Figure 2-5. The minimum DOC airplane corresponds to a wing loading of 120 lb/ft^2.

Figure 2-6 presents a weight breakdown comparison of the present and previous single stopped rotor aircraft.

Figure 2-7 shows the results of the parametric study of all four stopped rotor concepts. Both of the twin trailed rotor vehicles resulted in considerably higher gross weight aircraft with resultingly higher direct operating costs. For the single stowed rotor concept, the jet driven aircraft had a slightly lower gross weight and a higher cruise speed. The direct operating cost of the jet driven aircraft is, however, significantly higher. This is due entirely to the higher engine cost based directly on price quotes from the engine manufacturers. The single, stopped, stowed rotor aircraft driven by propellers for cruise flight is considered to be the best of this family of vehicles. Figure 2-8 presents a general arrangement of the stopped rotor aircraft.

FIGURE 2-3
SINGLE STOWED ROTOR - PROPELLER DRIVEN COMPARISON

	OLD (LR 19585)	NEW
GROSS WEIGHT (LB)	71,000	78,200
DOC (DOLLARS/SEAT MILE)	0.0245	0.288
BLOCK SPEED (KNOTS)	349	313
CRUISE VELOCITY (KNOTS)	425	402
CRUISE ALTITUDE (FT)	25,000	20,400
ROTOR TIP SPEED (FT/SEC)	800	800
MAIN ROTOR DIAMETER (FT)	95	119.2
DISK LOADING (LB/FT^2)	10	7
SOLIDITY OF MAIN ROTOR	0.0835	0.0598
ROTOR FIGURE OF MERIT	0.67	0.621
PROP DIAMETER (FT)	16	16
ACTIVITY FACTOR (PROPS)	160	140
PROPULSIVE EFF (CRUISE)	0.85	.85
RHP/ENGINE	4105	4290
WING LOADING (LB/FT^2)	120	120
WING AREA (FT^2)	592	656
WING SPAN (FT)	60	62.8
ASPECT RATIO	6	6

FIGURE 2-4

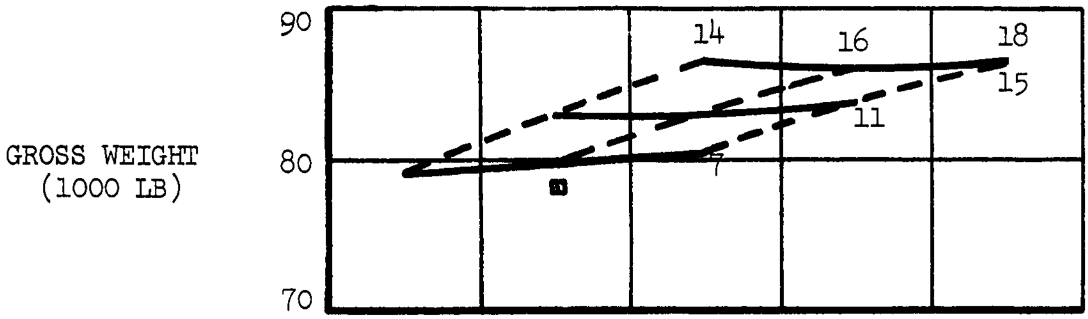
SINGLE STOWED ROTOR-PROPELLER DRIVEN
GROSS WEIGHTS AND D.O.C. FOR VARIOUS
PROPELLER DIAMETERS AND DISC LOADINGS

$W/S = 120$ LBS/FT^2

$V_T = 800$ FT/SEC

- - - - = P_D

______ = DL



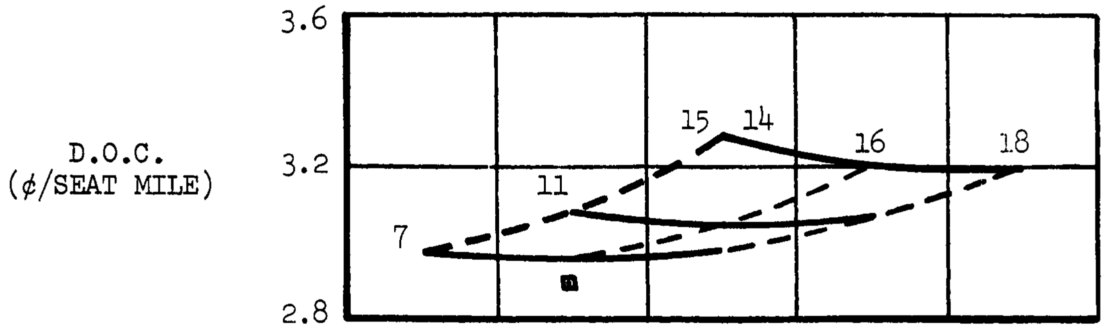


FIGURE 2-5

SINGLE STOWED ROTOR, PROPELLER DRIVEN

CHARACTERISTICS VARIATION WITH WING LOADING

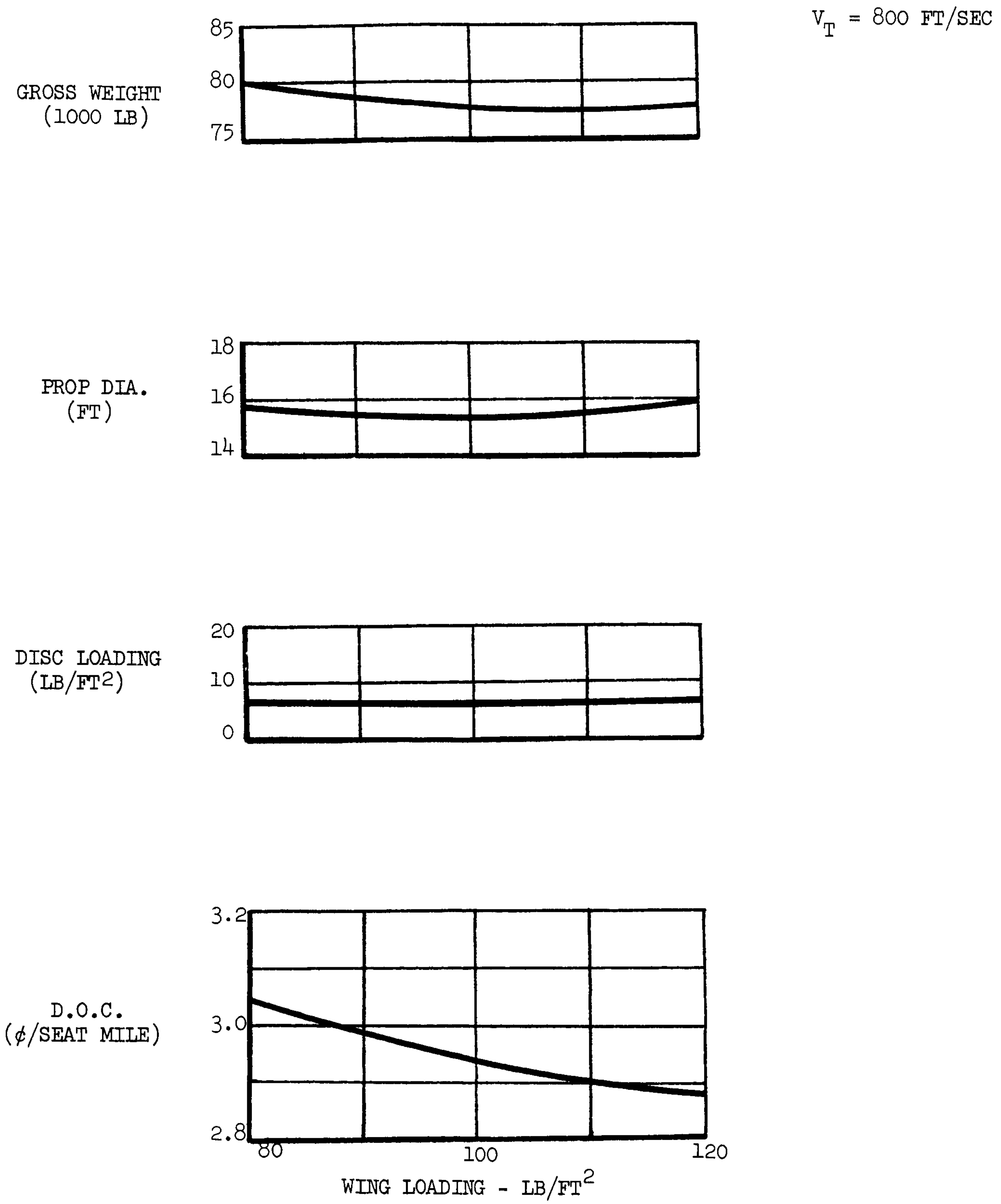


FIGURE 2-6

SINGLE STOWED ROTOR - WEIGHT STATEMENTS
(pounds)

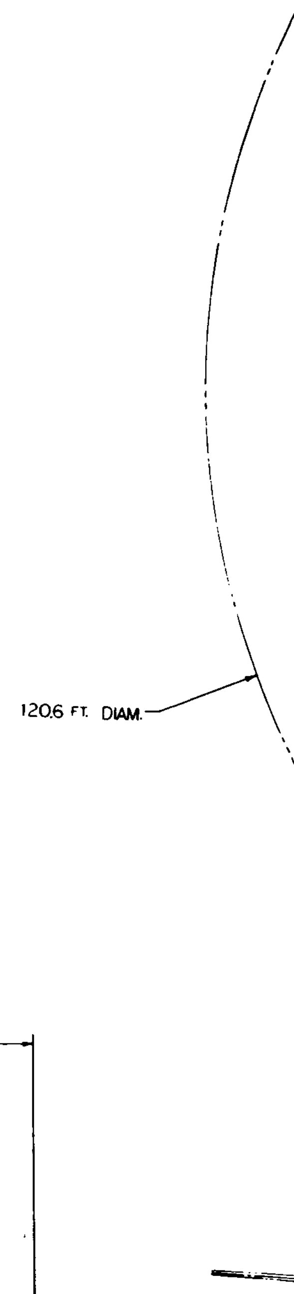
	OLD (LR 19585)		NEW	
	% W_G	WEIGHT	% W_G	WEIGHT
WING	5.04	3,580	5.12	4,005
TAIL	0.94	670	0.86	670
BODY	10.87	7,720	10.04	7,850
LANDING GEAR	3.79	2,690	3.84	3,000
FLIGHT CONTROLS	2.42	1,720	3.36	2,630
HYDRAULICS	0.51	360	0.47	370
INSTRUMENTS	0.59	420	0.52	405
AVIONICS	1.20	850	1.09	850
ELECTRICAL	1.27	900	1.21	950
AIR CONDITIONING	1.15	820	1.48	1,160
FURNISHINGS & EQUIPMENT	7.11	5,050	6.46	5,050
ANTI-ICING	0.62	440	0.63	490
AUXILIARY POWER UNIT	0.52	370	0.48	380
ENGINES	3.92	2,780	3.77	2,945
ENGINE ACCESSORIES	1.96	1,390	1.19	930
NACELLES	3.80	2,700	3.65	2,855
TAIL GEARBOX	0.52	370	0.39	305
MAIN GEARBOX	6.25	4,440	8.22	6,430
ENGINE GEARBOXES	0.37	250	0.35	275
CROSS SHAFT GEARBOXES	0.68	480	0.64	500
PROPELLER GEARBOXES	1.51	1,070	1.37	1,070
SHAFTING	0.56	400	0.61	475
ROTOR BRAKE AND CLUTCHES	0.70	500	0.70	550
PROPELLERS	2.65	1,880	1.79	1,400
MAIN ROTOR	10.00	7,100	11.25	8,800
TAIL ROTOR	0.93	660	1.15	900
FUEL SYSTEM	0.68	480	0.61	480
WEIGHT EMPTY	70.56	50,100	71.26	55,725
CREW	0.73	520	0.66	520
MISC. USEFUL LOAD	0.37	260	0.33	260
ENGINE OIL	0.27	195	0.26	205
UNUSABLE FUEL	0.18	125	0.16	125
OPERATING WEIGHT	72.11	51,200	72.68	56,835
PAYLOAD	18.59	13,200	16.88	13,200
ZERO FUEL WEIGHT	90.70	64,400	89.56	70,035
USABLE FUEL	9.30	6,600	10.44	8,165
GROSS WEIGHT	100.0	71,000	100.0	78,200

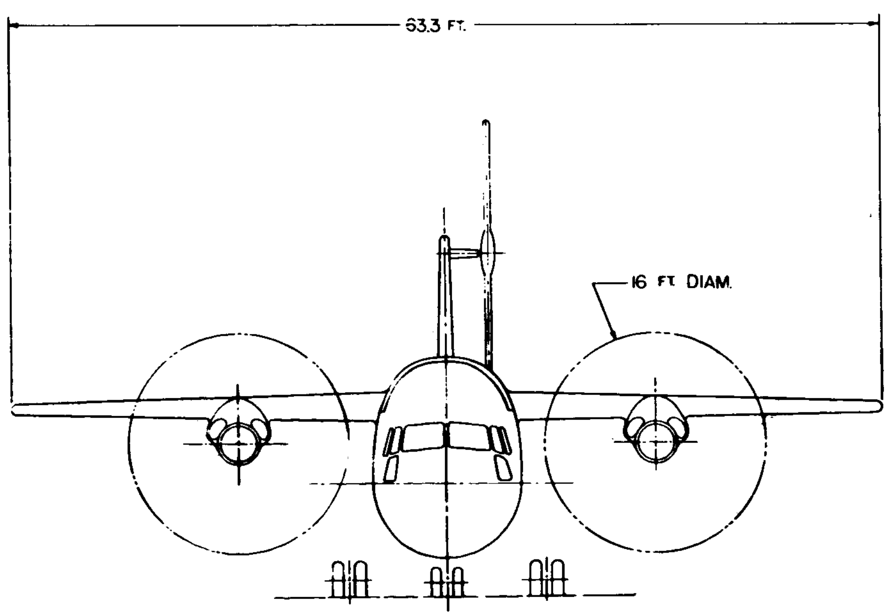
FIGURE 2-7

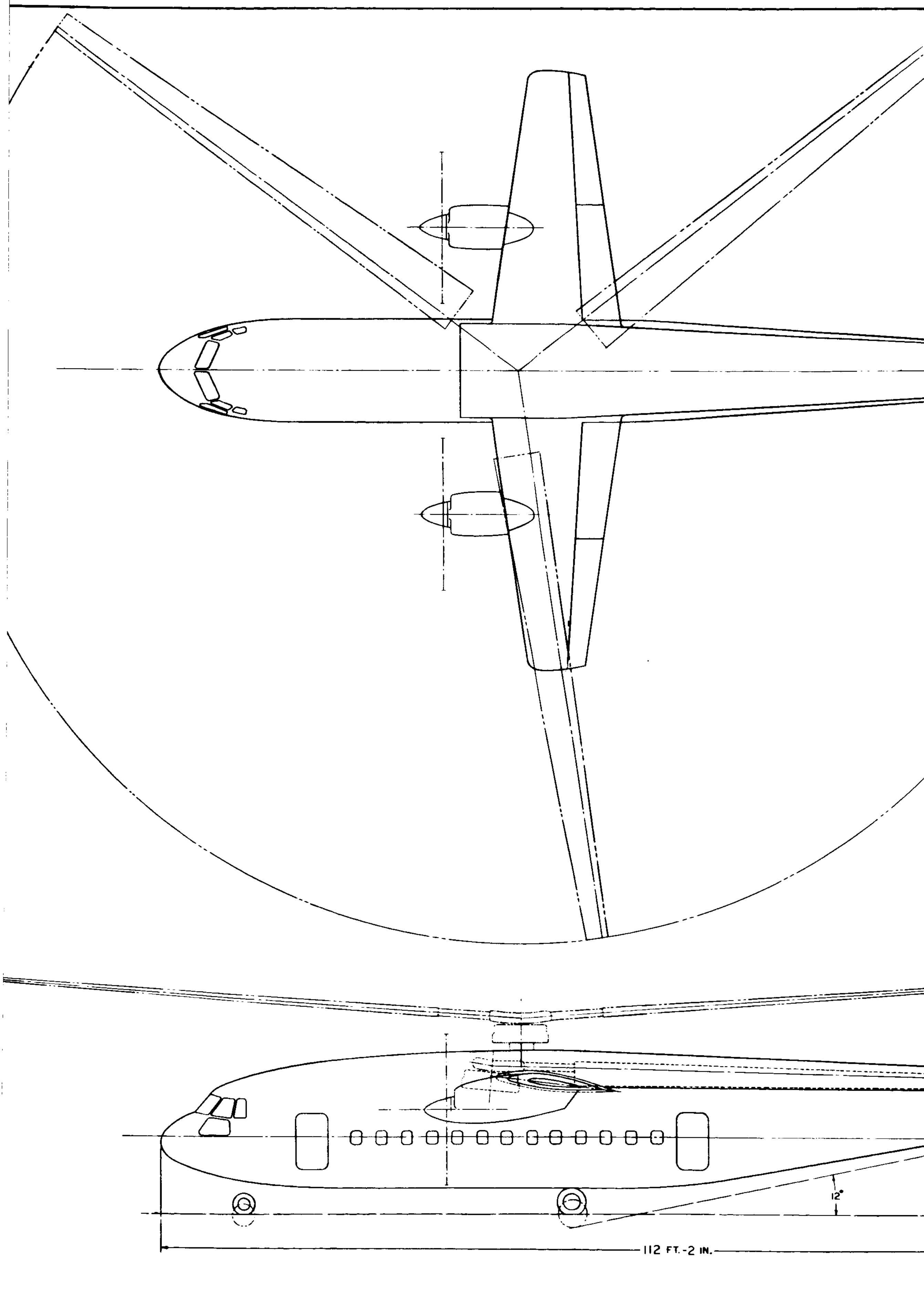
COMPARISON OF FOUR OPTIMIZED STOPPED ROTOR CONFIGURATIONS

PARAMETER	SINGLE ROTOR		TWIN ROTOR	
	PROP	JET	PROP	JET
GROSS WEIGHT (LBS)	78,200	76,600	87,700	84,500
WING AREA (FT^2)	656	696	643	867
WING SPAN (FT)	62.8	64.6	67.0	72.1
ASPECT RATIO	6	6	7	6
CRUISE VELOCITY (KNOTS)	402	438	399	435
CRUISE ALTITUDE (FT)	20,400	35,000	25,500	35,000
RHP/ENGINE	4290	4150	6886	5982
PROP ACTIVITY FACTOR	160	----	200	----
PROP DIAMETER (FT)	16	----	18	----
PROP TIP SPEED (FT/SEC)	800	----	700	----
ROTOR DIAMETER (FT)	119.2	118.1	65.1	70.1
SOLIDITY OF MAIN ROTOR	0.0598	0.0598	0.1138	0.0934
DISK LOADING (LBS/FT^2)	7	7	12.6	11
TAIL ROTOR RADIUS (FT)	9.56	9.48	----	----
ROTOR TIP SPEED (FT/SEC)	800	800	800	800
FIGURE OF MERIT	0.0621	0.621	0.621	0.621
DOC ($/SEAT MILE)	.0288	0.0312	0.0392	0.034

CHARACTERISTICS	WING	HORIZ.	VERT.
AREA ~ SQ. FT.	667	100	87
ASPECT RATIO	6	4	1.77
TAPER RATIO	.4	.5	.5
ENGINE	4-GE1/S1 4385 SHP EA.		
GROSS WEIGHT	79,900 LBS.		






12°
112 FT.-2 IN.

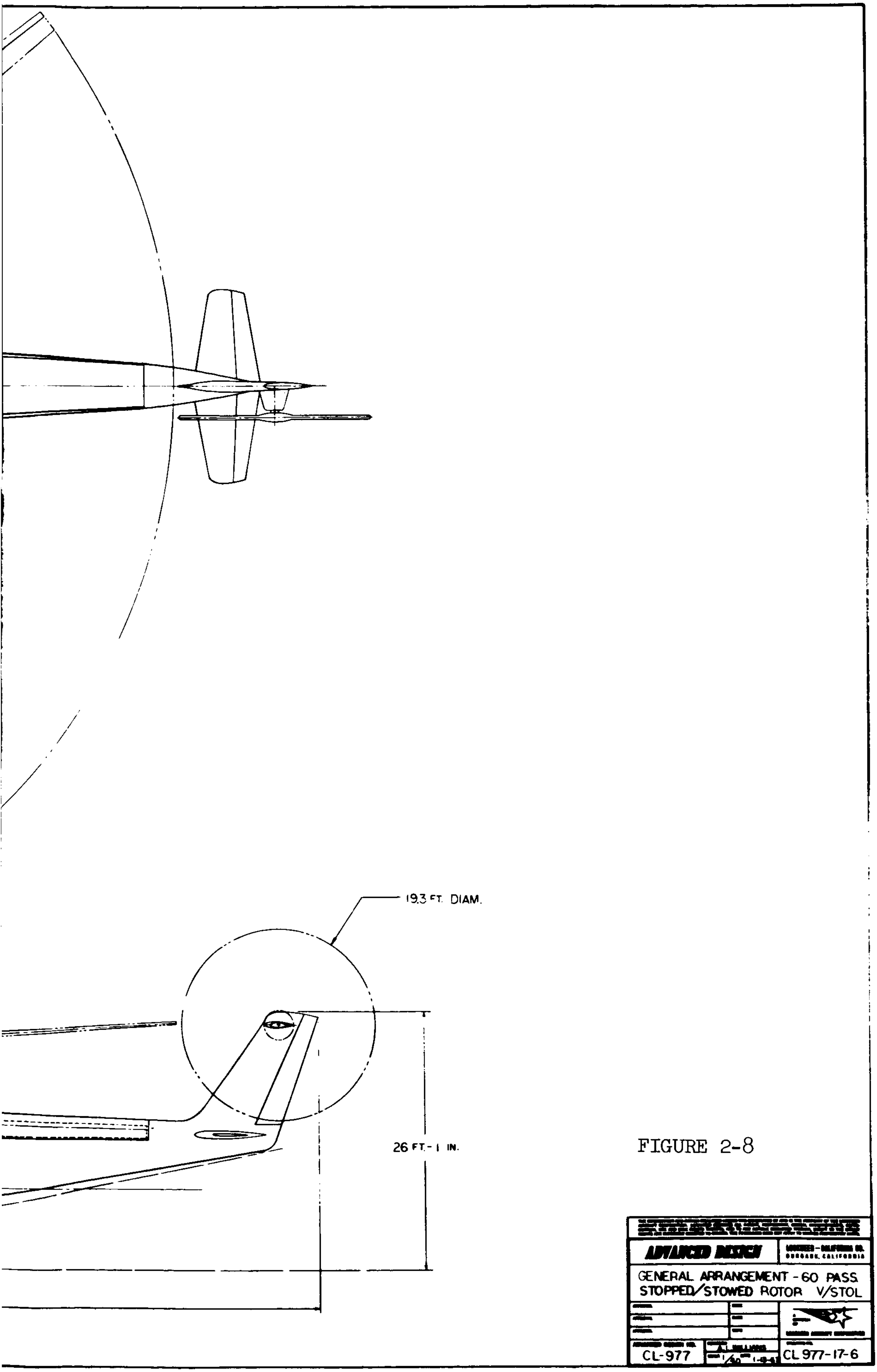

19.3 FT. DIAM.
26 FT.-1 IN.
ADVANCED DESIGN
GENERAL ARRANGEMENT - 60 PASS
STOPPED/STOWED ROTOR V/STOL
CL-977
CL 977-17-6

FIGURE 2-8

A comparison of the best tilt rotor aircraft from the current study and the comparable aircraft from the previously reported study is shown in Figure 2-9. Due to the considerably lower rotor figure of merit (.621 instead of .88) and the much more realistic cruise propeller efficiency (.765 instead of .96) the gross weight of the vehicle increases considerably. The optimum rotor tip speed is now found to be 800 ft/sec instead of 900 ft/sec due to the rapid drop in the figure of merit with increasing tip speed.

Figure 2-10 presents a typical set of results of the parametric study of the tilt rotor concept. The effect of rotor diameter and rotor tip speed on vehicle gross weight and direct operating cost are shown in this figure. As shown in the lower plot, the minimum DOC aircraft corresponds to a rotor tip speed of 800 feet per second and a rotor diameter of 66 feet. A weight breakdown comparison of the present and previous tilt rotor vehicles is shown in Figure 2-11.

Later studies indicate the possibility of obtaining a significantly higher rotor figure of merit at higher tip speeds by proper application of propeller technology to the design of rotor blades. Figure 2-12 shows this higher level curve compared to the present technology rotor curve used in the parametric study.

An additional examination was made of the effect on both the tilt and stopped rotor vehicles of the higher figure of merit. Figure 2-13 shows the results of this study applied to the tilt rotor aircraft. Both the gross weight and the direct operating cost were significantly lower. Due to the flatter characteristics of figure of merit with rotor tip speed, the optimum vehicle now has a rotor tip speed of 900 ft/sec. An increase of disc loading from 11.4 to 13 appears desirable. This increase, coupled with the lower gross weight, results in a considerably smaller rotor diameter.

The application of the same higher figure of merit values to the stopped rotor configurations provides the results shown in Figure 2-14. Again, both the gross weight and direct operating cost are lower than for the rotor technology vehicle. The optimum disc loading increases from 7 to 13 lb/ft^2 and the optimum tip speed increases to 900 ft/sec as was the case with the

FIGURE 2-9

TILT ROTOR COMPARISON

	OLD	NEW
GROSS WEIGHT (LBS)	60,800	77,900
DOC ($/SEAT MILE)	0.0235	0.0301
BLOCK SPEED (KNOTS)	313	311
CRUISE VELOCITY (KNOTS)	398	392
CRUISE ALTITUDE (FT)	27,500	35,000
ROTOR TIP SPEED (FT/SEC)	900	800
PROP DIAMETER (ROTOR) (FT)	44.83	66.0
DISK LOADING (LB/FT^2)	15.55	11.4
SOLIDITY (MAIN ROTOR)	0.085	0.095
ACTIVITY FACTOR	52.2	58.3
FIGURE OF MERIT (ROTOR)	0.88	0.621
PROPULSIVE EFFICIENCY (CRUISE)	0.96	0.765
RHP/ENGINE	3040	5580
WING LOADING (LB/FT^2)	88.0	72.0
WING AREA (FT^2)	690.0	1077.0
WING SPAN (FT)	64.0	80.4
ASPECT RATIO	6.0	6.0

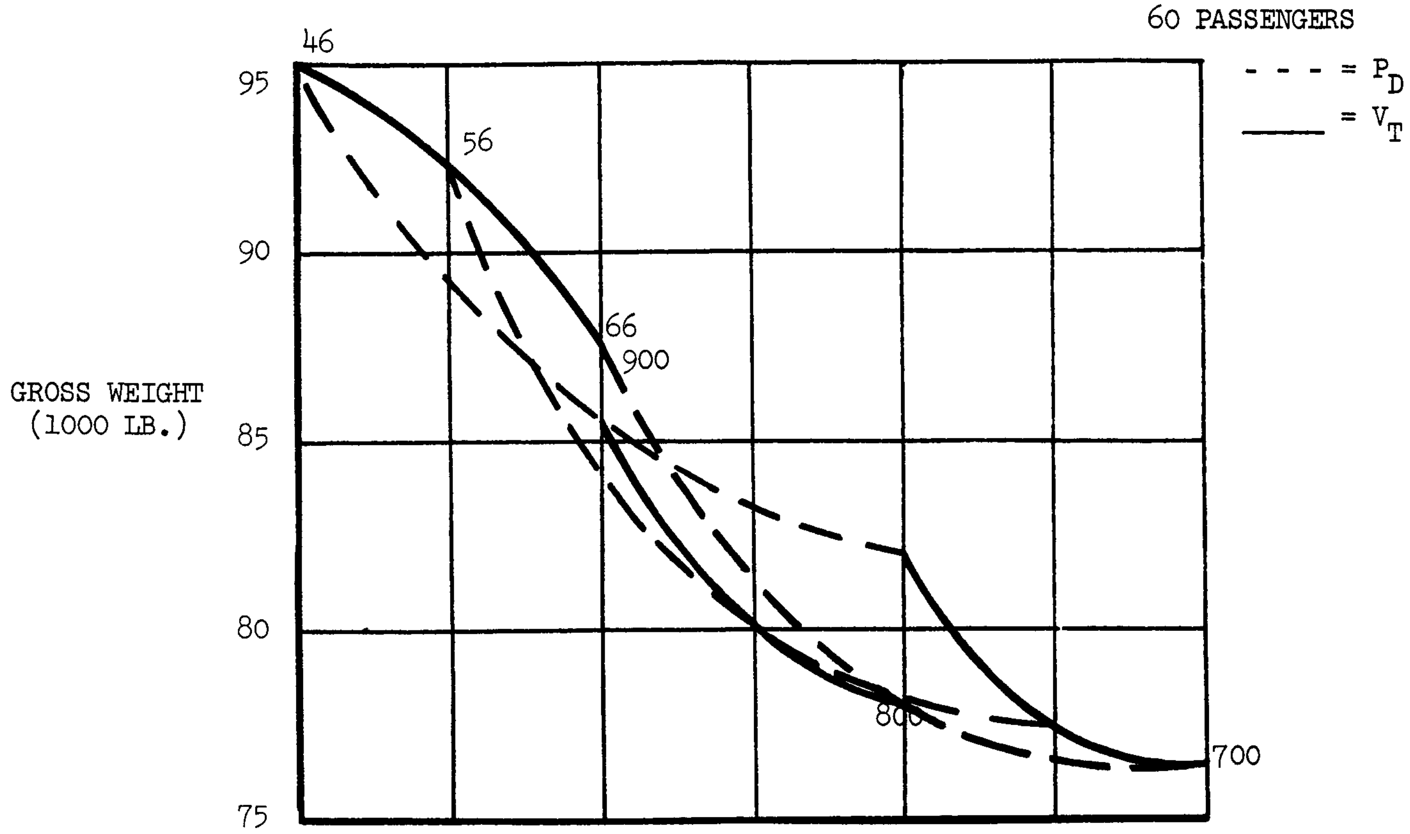
FIGURE 2-10

TILT ROTOR

GROSS WT. AND D.O.C. FOR VARIOUS

PROPELLER DIAMETERS AND TIP SPEEDS

RANGE = 500 ST. MILES
AR = 6
60 PASSENGERS



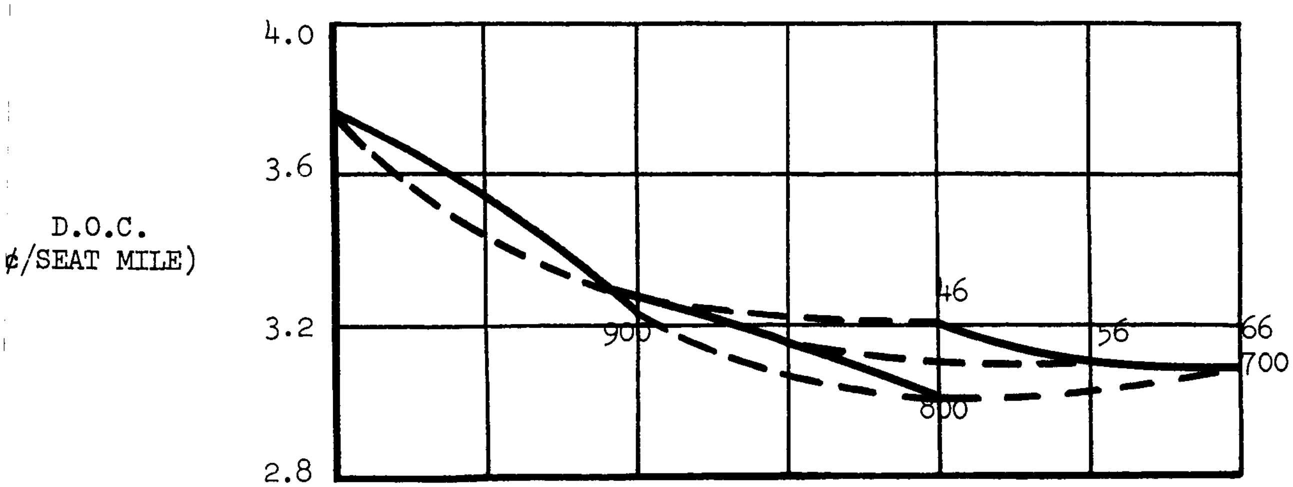


FIGURE 2-11

TILT ROTOR - WEIGHT STATEMENTS
(pounds)

	OLD (LR 19585)		NEW	
	% W_G	WEIGHT	% W_G	WEIGHT
WING	7.12	4,330	8.58	6,685
EMPENNAGE	2.52	1,530	2.00	1,555
FUSELAGE	10.38	6,310	8.80	6,855
LANDING GEAR	3.78	2,300	3.82	2,980
FLIGHT CONTROLS	3.70	2,250	3.64	2,835
HYDRAULICS	0.62	380	0.56	435
INSTRUMENTS	0.67	410	0.56	435
ELECTRICAL	1.56	950	1.22	950
AVIONICS	1.40	850	1.09	850
AIR CONDITIONING	1.91	1,160	1.49	1,160
FURNISHINGS & EQUIPMENT	8.24	5,010	6.55	5,100
ANTI-ICING	0.76	465	0.72	560
AUXILIARY POWER UNIT	0.59	360	0.49	380
ENGINES	3.68	2,240	4.60	3,585
ENGINE ACCESSORIES	1.84	1,120	1.44	1,120
NACELLES	3.57	2,170	4.47	3,480
MAIN ROTORS	7.97	4,845	10.06	7,840
MAIN GEARBOXES	5.77	3,510	8.66	6,745
CROSS SHAFT GEARBOXES	0.51	310	0.64	495
ENGINE GEARBOXES	0.35	210	0.42	330
CROSS SHAFTING	0.49	295	0.62	480
CLUTCHES	0.28	170	0.39	305
FUEL SYSTEM	0.74	450	0.62	480
WEIGHT EMPTY	68.46	41,625	71.42	55,640
CREW	0.85	520	0.67	520
MISC. USEFUL LOAD	0.43	260	0.33	260
ENGINE OIL	0.25	150	0.35	270
UNUSABLE FUEL	0.12	75	0.15	120
OPERATING WEIGHT	70.11	42,630	72.92	56,810
PAYLOAD	21.71	13,200	16.95	13,200
ZERO FUEL WEIGHT	91.82	55,830	89.87	70,010
USABLE FUEL	8.18	4,970	10.13	7,890
GROSS WEIGHT	100.0	60,800	100.0	77,900

FIGURE 2-12

FIGURE OF MERIT VS. ROTOR TIP SPEED

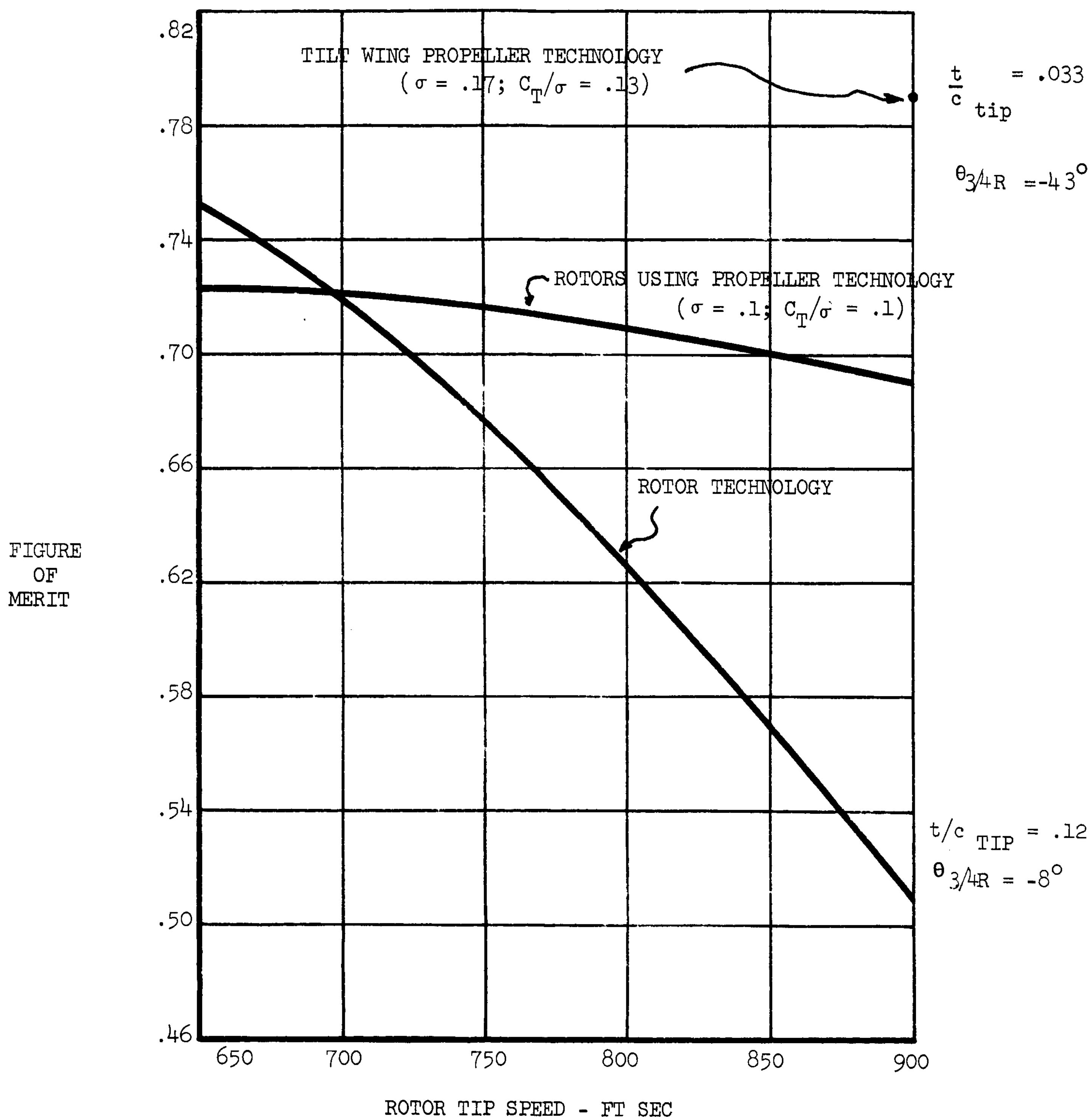


FIGURE 2-13

COMPARISON OF TILT ROTOR CHARACTERISTICS
PROPELLER VS ROTOR TECHNOLOGY

	PROP	ROTOR
GROSS WEIGHT (LB)	65,000	77,900
DOC ($/SEAT MILE)	0.0270	0.0301
BLOCK SPEED (KNOTS)	296	311
CRUISE VELOCITY (KNOTS)	363	392
CRUISE ALTITUDE (FT)	25,000	35,000
ROTOR TIP SPEED (FT/SEC)	900	800
PROP DIAMETER (ROTOR) (FT)	56.4	66.0
DISK LOADING (LB/FT^2)	13	11.4
SOLIDITY (MAIN ROTOR)	0.086	0.095
ACTIVITY FACTOR	52.6	58.3
FIGURE OF MERIT (ROTOR)	0.69	0.621
PROPULSIVE EFFICIENCY (CRUISE)	0.765	0.765
RHP/ENGINE	3840	5580
WING LOADING (LB/FT^2)	77.8	72
WING AREA (FT^2)	835	1077
WING SPAN (FT)	70.8	80.4
ASPECT RATIO	6	6

FIGURE 2-14

COMPARISON OF STOWED ROTOR CHARACTERISTICS
PROPELLER VS ROTOR TECHNOLOGY

	Prop Technology	Rotor Technology
Gross Weight (lb)	71,000	78,200
DOC (dollars/seat mile)	0.0265	0.0288
Block Speed (knots)	312	313
Cruise Velocity (knots)	400	402
Cruise Altitude (ft)	20,000	20,400
Rotor Tip Speed (ft/sec)	900	800
Main Rotor Diameter (ft)	83.4	119.2
Disk Loading (lb/ft^2)	13	7
Solidity of Main Rotor	0.0878	0.0598
Rotor Figure of Merit	0.69	0.621
Prop Diameter (ft)	16	16
Activity Factor (props)	140	140
Propulsive EFF (cruise)	0.85	0.85
RHP/Engine	4350	4290
Wing Loading (lb/ft^2)	120	120
Wing Area (ft^2)	592	656
Wing Span (ft)	60	62.8
Aspect Ratio	6	6

tilt rotor. This results in a much smaller rotor than for the case with the rotor technology aircraft.

The weight statements for the 60-passenger tilt and stopped rotor aircraft utilizing propeller technology rotor blades are shown in Figure 2-15.

A 120-passenger tilt rotor aircraft was also weighed, performed, and costed. The weight statement for this aircraft is shown in Figure 2-16.

The aerodynamics, weight, propulsion, and cost methods used in this study are identical to those previously detailed in Addendum One, LR 19585, Volumes I, II, and III with the following exceptions.

Additional propeller performance for the tilt rotor configuration in the cruise mode was calculated using the Hamilton Standard strip analysis propeller program. Typical results of this program are shown in Figures 2-17 through 2-20. A range of activity factors from 35 to 200 was considered for various tip speeds and rotor diameters. The results of this series of runs were used in selection of a rotor to be combined with basic engine data to obtain installed thrust and fuel flow characteristics for the tilt rotor vehicles.

The weights of propellers and propeller gearboxes have been reduced 15 percent due to revised Hamilton Standard propeller data. Engine accessories weight has also been revised to incorporate later input data. The revised equation is:

$$W_{EA} = (.785 \text{ RHP})^{.843}$$

W_{EA} - Engine accessories weight

RHP - Rated horsepower/engine (4 engines)

Rotor weights have been revised slightly for propeller technology rotors.

The engine data used is the same as previously detailed in LR 19585 except that the stopped rotor fan versions utilize Allison 902-H4 fan shaft engines and accompanying data.

FIGURE 2-15

STOPPED ROTOR AND TILT ROTOR WEIGHT STATEMENTS - PROPELLER TECHNOLOGY
(pounds)

	TILT ROTOR	STOPPED ROTOR
WING	4,755	3,600
TAIL	1,275	610
BODY	6,320	7,010
LANDING GEAR	2,390	2,720
FLIGHT CONTROLS	2,580	2,440
HYDRAULICS	405	360
INSTRUMENTS	410	405
AVIONICS	850	850
ELECTRICAL	950	950
AIR CONDITIONING	1,160	1,160
FURNISHINGS & EQUIPMENT	5,040	5,060
ANTI-ICING	500	440
AUX. POWER UNIT	365	370
ENGINES	2,680	2,970
ENGINE ACCESSORIES	820	920
NACELLES	2,600	2,880
TAIL GEARBOX	---	300
MAIN GEARBOXES	4,030	4,460
ENGINE GEARBOXES	250	275
CROSS SHAFT GEARBOXES	370	510
PROPELLER GEARBOXES	---	850
SHAFTING	380	390
ROTOR BRAKE & CLUTCHES	220	495
PROPELLERS	---	1,390
MAIN ROTORS	5,520	6,100
TAIL ROTOR	---	755
FUEL SYSTEM	465	480
WEIGHT EMPTY	44,335	48,750
CREW	520	520
MISC. USEFUL LOAD	260	260
ENGINE OIL	185	210
UNUSABLE FUEL	100	120
OPERATING WEIGHT	45,400	49,860
PAYLOAD	13,200	13,200
ZERO FUEL WEIGHT	58,600	63,060
USABLE FUEL	6,400	7,940
GROSS WEIGHT	65,000 lb	71,000 lb

FIGURE 2-16
WEIGHT STATEMENT - 120 PASSENGER TILT ROTOR
(pounds)

WING	11,080
TAIL	2,680
BODY	12,750
LANDING GEAR	4,870
FLIGHT CONTROLS	3,510
HYDRAULICS	540
INSTRUMENTS	470
ELECTRICAL	1,800
AVOINICS	1,100
AIR CONDITIONING	2,220
FURNISHINGS & EQUIPMENT	9,330
ANTI-ICING	800
AUXILIARY POWER UNIT	420
ENGINES	4,385
ENGINE ACCESSORIES	1,395
NACELLES	4,255
MAIN ROTORS	11,200
MAIN GEARBOXES	8,420
CROSS SHAFT GEARBOXES	600
ENGINE GEARBOXES	405
CROSS SHAFTING	640
CLUTCHES	390
FUEL SYSTEM	515
WEIGHT EMPTY	83,775
CREW	660
MISC. USEFUL LOAD	510
ENGINE OIL	340
UNUSABLE FUEL	175
OPERATING WEIGHT	85,460
PAYLOAD	26,400
ZERO FUEL WEIGHT	111,860
USABLE FUEL	11,640
GROSS WEIGHT	123,500 pounds

FIGURE 2-17
PROPELLER EFFICIENCY VS. MACH NUMBER
FOR VARIOUS RATED HORSEPOWERS

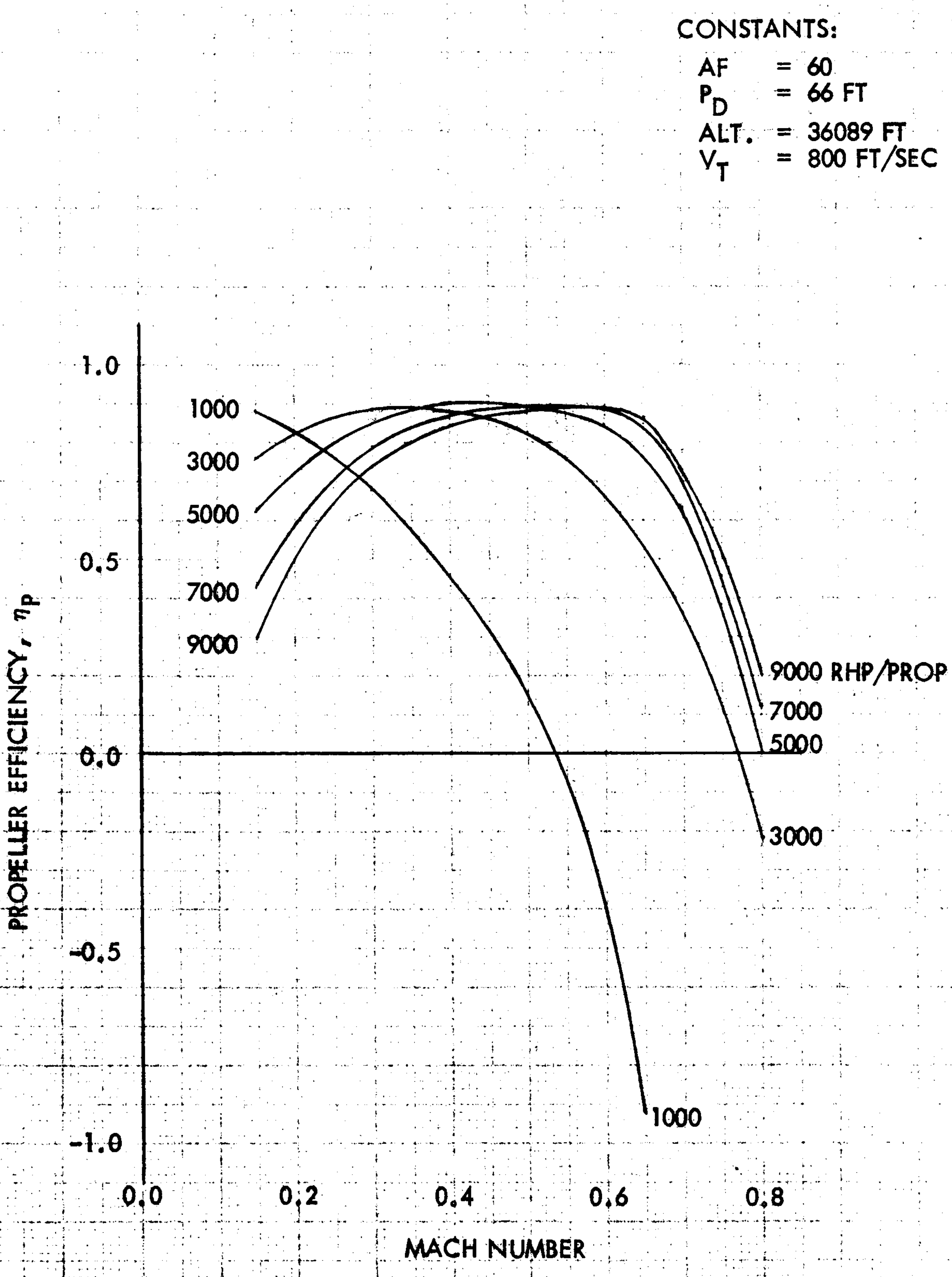


FIGURE 2-18
PROPELLER EFFICIENCY VS. MACH NUMBER
FOR VARIOUS RATED HORSEPOWERS

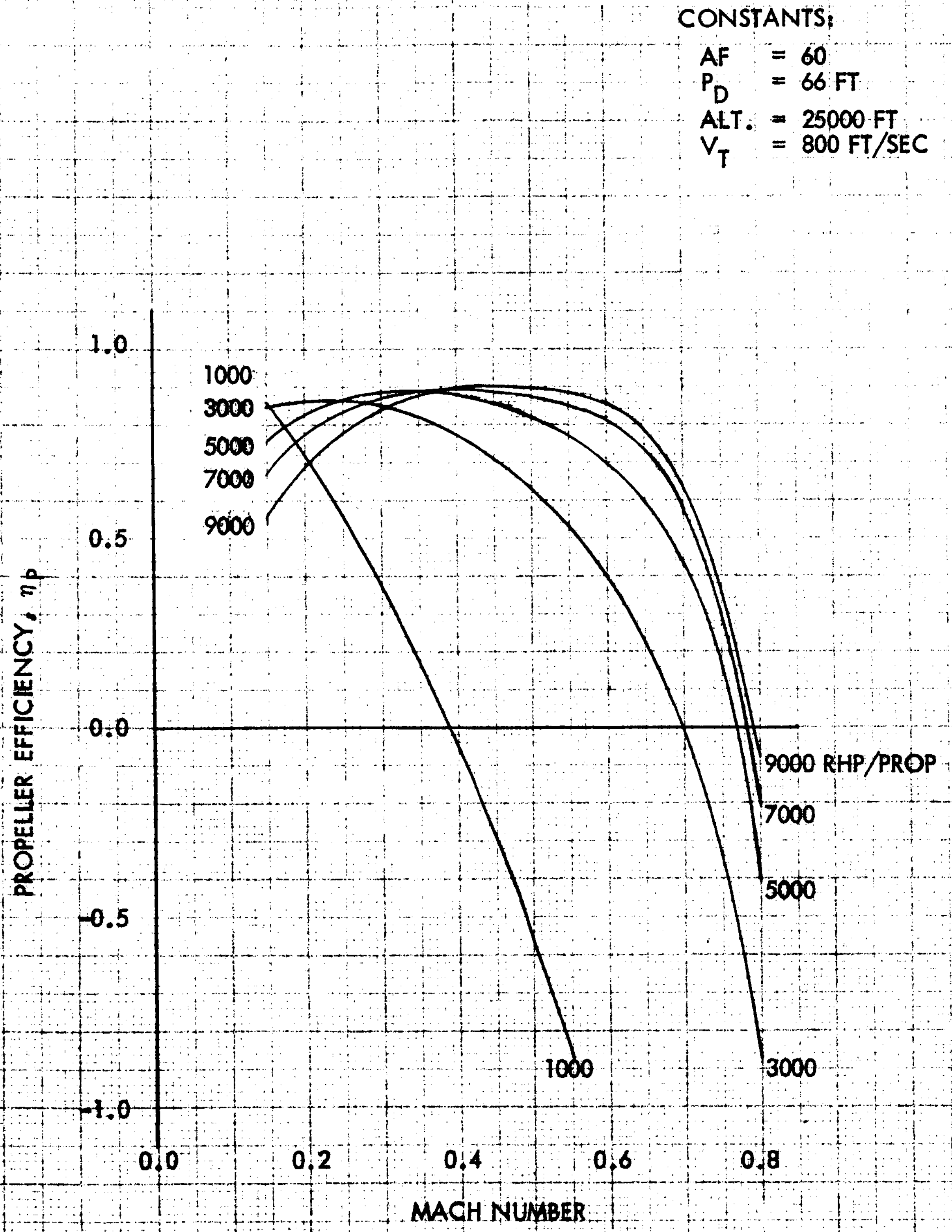


FIGURE 2-19
PROPELLER EFFICIENCY VS. MACH NUMBER
FOR VARIOUS RATED HORSEPOWERS

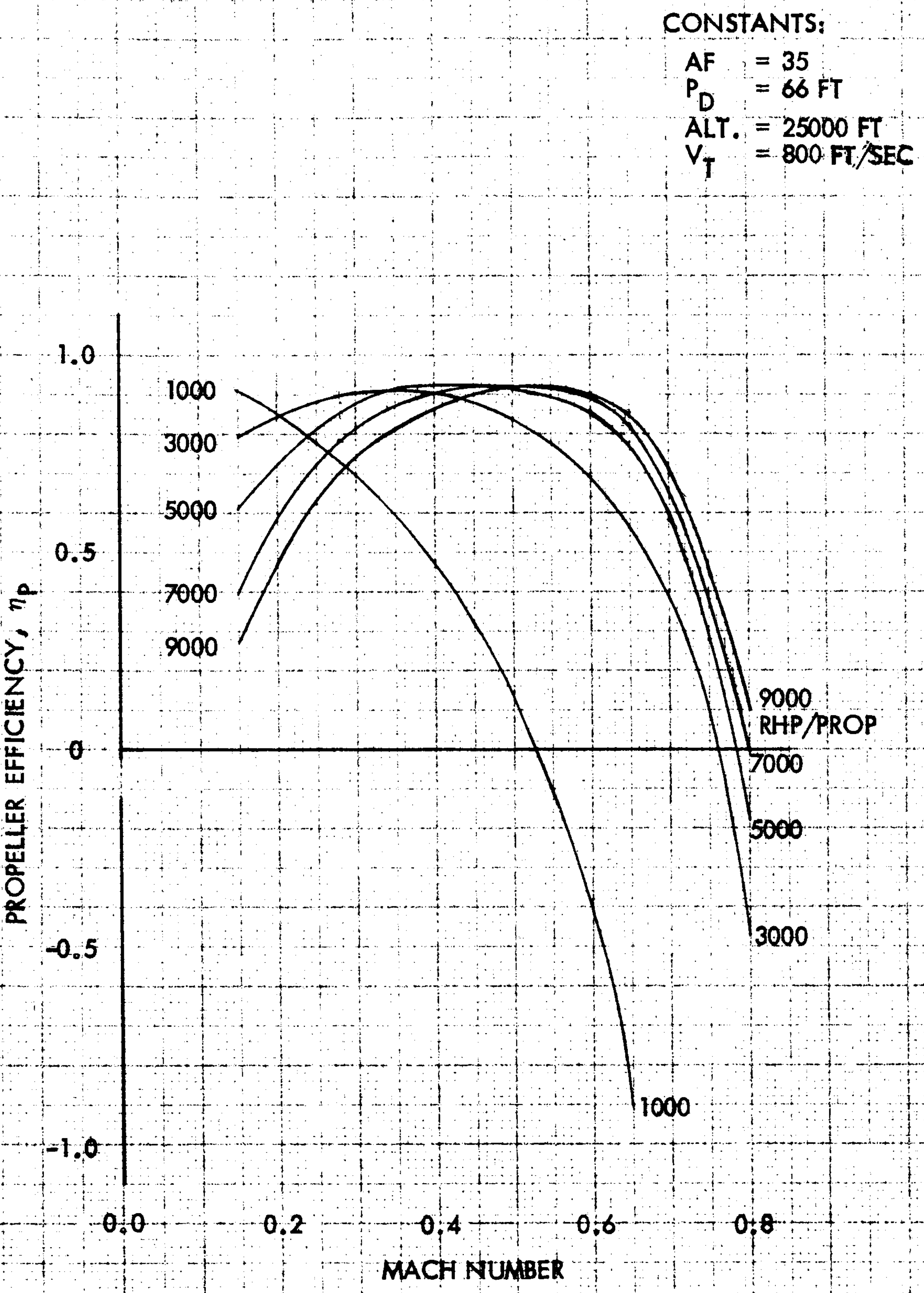
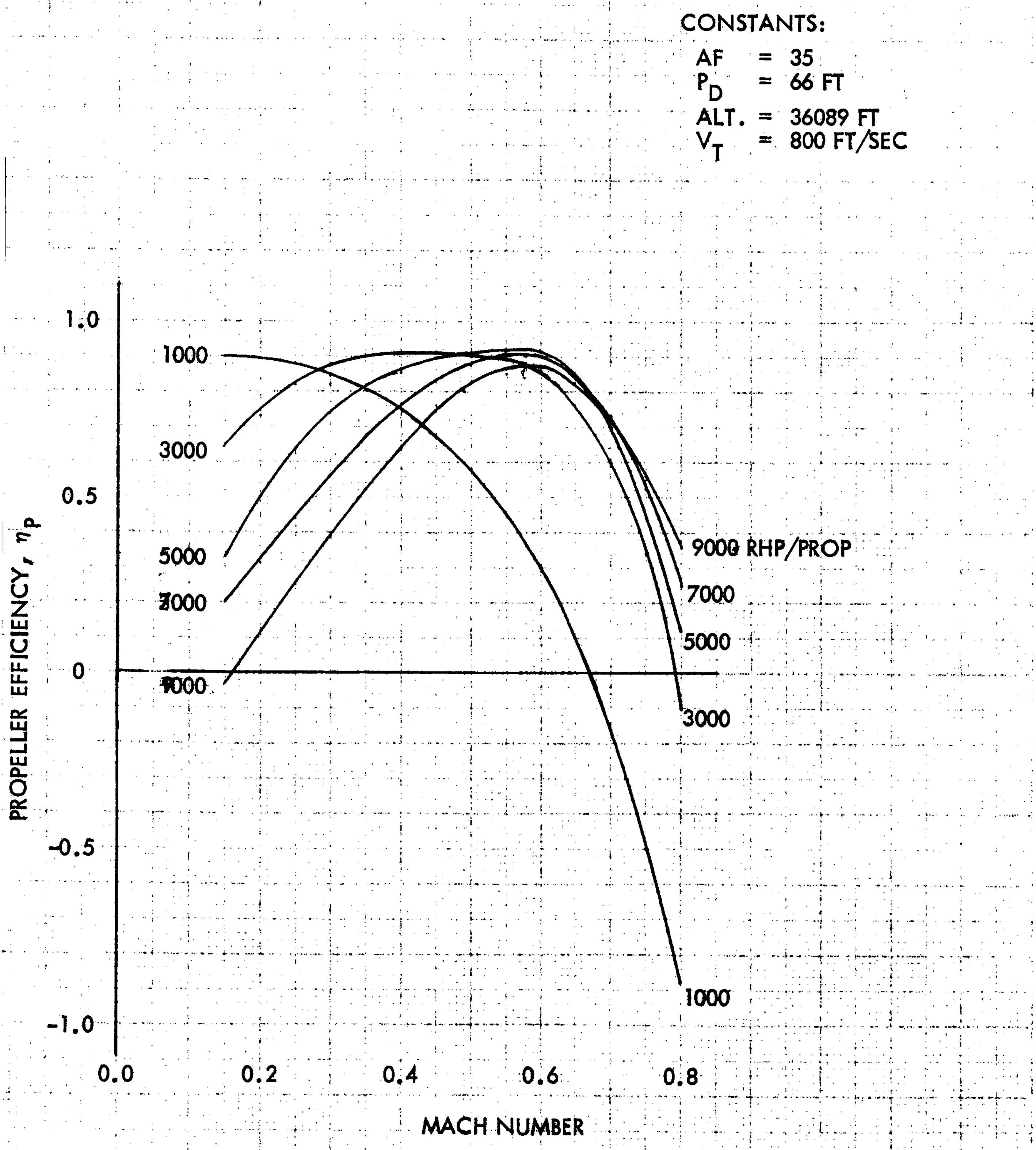


FIGURE 2-20
PROPELLER EFFICIENCY VS. MACH NUMBER
FOR VARIOUS RATED HORSEPOWERS



The cost model used to generate the direct operating costs for the aircraft in this report is the same as the cost model presented in LR 19585 except for a new engine equation used to calculate the flyaway costs of the stopped rotor fan driven aircraft, and for new rotor and gearbox maintenance equations. The following engine cost equation was derived from data provided by the Allison Company:

$$C_e = 681234 \left[1.0 + .4 \frac{RFN-8170}{8170}\right] Q_{eng}^{-.152}$$

Where:

C_e = Cost per engine in dollars

RFN = Maximum engine thrust (S.L.S.) in pounds

Q_{eng} = Quantity produced for the total program

The new rotor and gearbox maintenance equation for tilt and stopped rotor aircraft is:

$$\text{Rotor and Gearbox Maintenance} = \frac{.000207 \ WG + 15.6 \ WGB + NR \ (648.6) \ DR \ (OT)^{.25}}{V_b}$$

Where:

WG = Aircraft gross weight in pounds

WGB = Gearbox and shaft weight in pounds

NR = Number of rotors

DR = Diameter of rotor in inches

OT = Rotor operating time per flight in hours

V_b = Block speed in miles per hour

The rotor and gearbox maintenance cost equation for the stowed rotor aircraft is identical to the equation for the tilt and stopped rotor except that the whole quantity is increased by 7.5 percent to account for maintenance of the stowing mechanism.

3. NOISE SENSITIVITY ANALYSIS

During the Short Haul Transport Study it became evident that noise is a major problem for all short-haul aircraft. Therefore a study was conducted to assess the sensitivity of far-field perceived noise to parametric changes in aircraft design in terms of weight, speed, and DOC.

In order to evaluate the sensitivity of far-field noise to aircraft design changes, the propeller and/or rotor speed was varied on the Deflected Slipstream, Tilt Rotor, and Stopped Rotor concepts. Aircraft were designed for tip speeds of 700, 800, and 900 fps. For the Fan-In-Wing and Jet Flap concepts, far-field noise was determined as a function of T/W ratio for values corresponding to 1000-ft and 2000-ft field lengths.

The physical characteristics of the 60-passenger aircraft selected for noise sensitivity analysis are tabulated in Figure 3-1. The Deflected Slipstream aircraft are 2000-ft STOL vehicles. Therefore W/S and T/W ratios are held constant as propeller tip speed is varied. The tip speed variation affects the propeller activity factor selection and the engine power requirements. The Jet Flap and Fan-In-Wing aircraft were designed for two field lengths of 1000-ft and 2000-ft. This results in significant changes in gross weight, engine power, T/W, and tail areas. The tilt rotor and stopped rotor are VTOL aircraft. The tip speed variation affects figure of merit or engine power requirements, rotor blade characteristics, and gearbox torque requirements. These variations affect the vehicle gross weights.

The 500 statute mile range performance for the aircraft selected for noise sensitivity analysis are shown in Figure 3-2.

To determine the effects on noise, a two point evaluation was selected, one for the aircraft in an on-ground condition, the other for a fly-over condition. The fly-over conditions are shown in Figure 3-3. The aircraft and engine performance data, at the two locations selected for the evaluation, were used to calculate the noise for each aircraft.

FIGURE 3-1A

PHYSICAL CHARACTERISTICS OF SELECTED AIRCRAFT FOR NOISE SENSITIVITY ANALYSIS

AIRCRAFT TYPE	NOISE MODEL	Wg (pounds)	AR	λ	t/c Root	t/c Tip	S (sq ft)	W/S (lb/sq ft)	b (feet)	$\Lambda c/4$ (degrees)
Deflected Slipstream (2000-ft STOL)	900 fps	46,900	6	.70	.15	.13	832	56	71	0
	800 fps	48,000	6	.70	.15	.13	857	56	72	0
	700 fps	49,300	6	.70	.15	.13	876	56	73	0
Jet Flap	1000-ft STOL	77,700	8	.40	.15	.13	971	80	88	25
	2000-ft STOL	63,200	8	.40	.13	.10	843	75	82	25
Fan-In-Wing	1000-ft STOL	67,900	6	.44	.13	.11	1069	64	80	25
	2000-ft STOL	59,000	6	.44	.13	.11	880	67	73	25
Tilt Rotor (VTOL)	900 fps	65,000	6	.60	.16	.14	835	78	71	0
	800 fps	67,900	6	.60	.16	.14	890	76	73	0
	700 fps	76,400	6	.60	.16	.14	1067	72	80	0
Stopped Rotor (VTOL)	900 fps	71,000	6	.60	.14	.12	592	120	60	0
	800 fps	75,000	6	.60	.14	.12	625	120	61	0
	700 fps	85,700	6	.60	.14	.12	714	120	65	0

FIGURE 3-1B

PHYSICAL CHARACTERISTICS OF SELECTED AIRCRAFT FOR NOISE SENSITIVITY ANALYSIS

AIRCRAFT TYPE	NOISE MODEL	P_D (feet)	W/A (lb/sq ft)	T/W	T/Eng. (pounds)	SHP/Eng.	F_D (lift) (inches)	F_D (cruise) (inches)	S_H (sq ft)	S_V (sq ft)
Deflected Slipstream (2000-ft STOL)	900 fps	14	-	.47	-	1275	-	-	237	211
	800 fps	14	-	.47	-	1390	-	-	237	211
	700 fps	14	-	.47	-	1610	-	-	237	211
Jet Flap	1000-ft STOL	-	-	.60	12,640	-	-	-	216	249
	2000-ft STOL	-	-	.40	6,800	-	-	-	125	172
Fan-In-Wing	1000-ft STOL	-	-	.35	6,488	-	55	-	330	293
	2000-ft STOL	-	-	.28	4,370	-	49	-	243	247
Tilt Rotor (VTOL)	900 fps	56	13	-	-	3840	-	-	237	122
	800 fps	58	13	-	-	3980	-	-	248	127
	700 fps	66	11.2	-	-	4700	-	-	278	143
Stopped Rotor (VTOL)	900 fps	16[a]	13	-	-	4350	-	-	63	104
	800 fps	16[b]	13	-	-	4650	-	-	67	110
	700 fps	16[c]	13.6	-	-	5690	-	-	76	126

[a] Rotor Diameter 83.4 ft
[b] Rotor Diameter 85.6 ft
[c] Rotor Diameter 89.5 ft

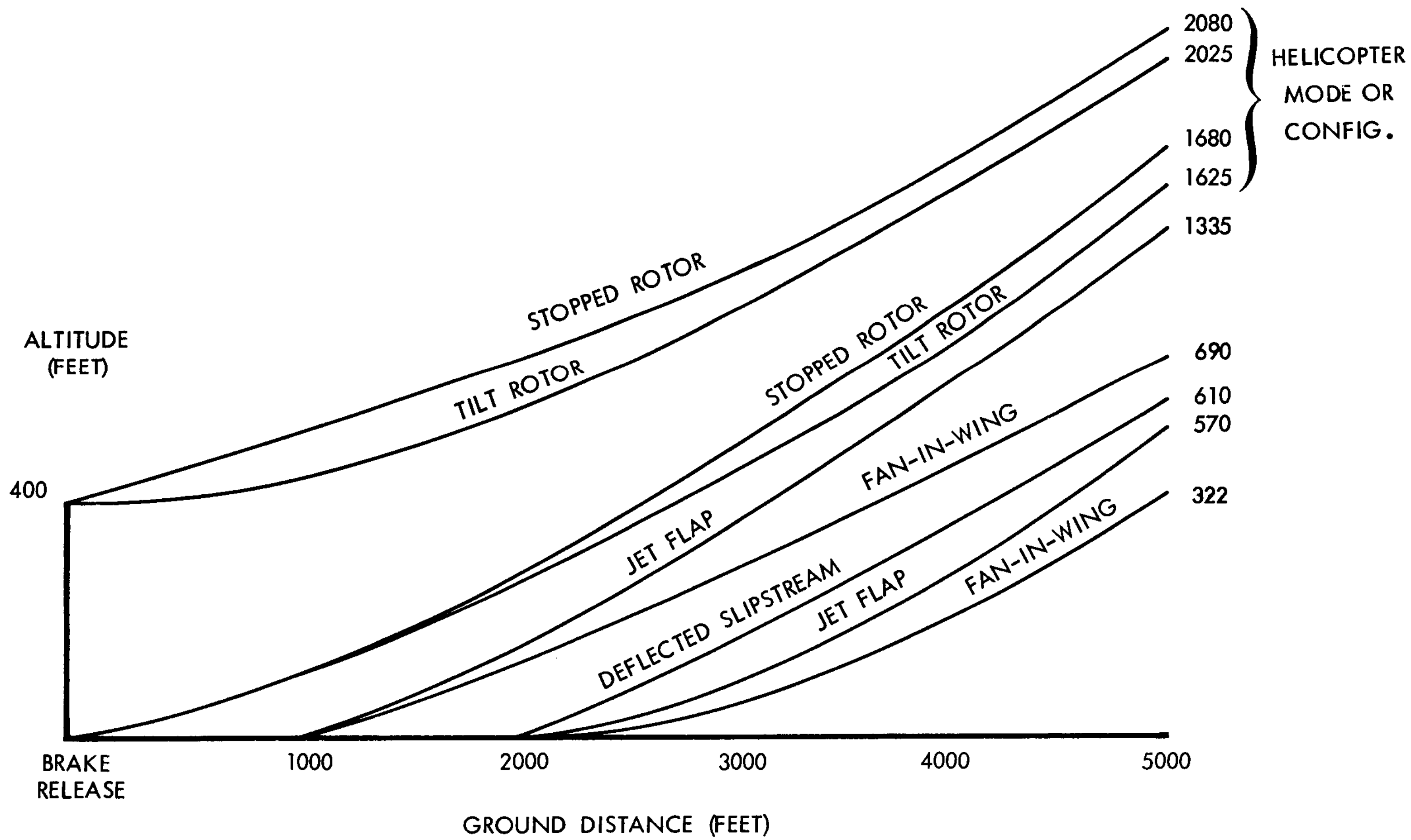
FIGURE 3-2

500 STATUTE MILE RANGE PERFORMANCE FOR SELECTED AIRCRAFT FOR NOISE SENSITIVITY ANALYSIS

AIRCRAFT TYPE	NOISE MODEL	W_f (pounds)	W_f Block (pounds)	V_{Cruise} (knots)	V_{Block} (mph)	η_p (cruise)	Fig. of Merit	H_p Cruise (feet)	D.O.C. ¢/seat mi.
Deflected Slipstream (2000-ft STOL)	900 fps	3220	2408	283	281	.92	-	15,280	1.96
	800 fps	3510	2680	298	311	.90	-	15,000	1.90
	700 fps	3780	2920	335	323	.89	-	15,000	1.86
Jet Flap	1000-ft STOL	14,440	10,510	478	406	-	-	31,000	2.90
	2000-ft STOL	9875	7294	483	424	-	-	31,000	2.26
Fan-In-Wing	1000-ft STOL	13,980	10,640	493	440	-	-	30,000	2.67
	2000-ft STOL	10,700	7880	475	419	-	-	30,000	2.48
Tilt Rotor (VTOL)	900 fps	6400	4350	363	341	.765	.69	25,000	2.67
	800 fps	6680	4540	363	341	.78	.71	25,000	2.70
	700 fps	7500	5100	365	335	.79	.72	25,000	3.09
Stopped Rotor (VTOL)	900 fps	7940	6260	400	359	.85	.69	20,000	2.65
	800 fps	8400	6620	405	361	.85	.71	20,000	2.70
	700 fps	9600	7560	437	370	.85	.72	20,000	3.12

FIGURE 3-3

TAKEOFF AND CLIMB PROFILES



Noise environments are usually described in terms of sound pressure level (SPL), a readily measurable quantity, which is defined as: SPL = 20 $\log_{10}\left(\frac{p}{p_r}\right)$ where $\underline{p}$ is the r.m.s. pressure fluctuations (in dynes/sq. cm.) and $\underline{p_r}$ is the reference pressure (0.0002 dynes/sq.cm. - the threshold of hearing at 1000 Hz). The units of SPL are decibels (dB). In a similar manner, the total acoustic power radiated by a noise source is described by the sound power level (PWL) which is defined as: PWL = 10 $\log_{10}\left(W/W_r\right)$ where $\underline{W}$ is rms power radiated (in watts) and $\underline{W_r}$ is the reference power (10^{-13} watts). The PWL is also expressed in dB. The relationship between SPL and PWL for spherical spreading is:

$$SPL = PWL + DI - 20 \log s - 10.5$$

Where:

SPL = sound pressure level (dB re 0.0002 dyne/sq.cm.)
PWL = sound power level (dB re 10^{-13} watt)
DI = directivity index (dB re space average PWL)
s = radial distance (ft)

One measure of the "noisiness" or annoyance of a sound, commonly used in aircraft work, is the perceived noise level (PNL) expressed in units of PNdB (perceived noise decibels). The PNL is derived from subjective tests and relates the noisiness of a broad band noise to an equivalent noisiness of a band of noise centered at 1000 Hz. The PNL is a computed quantity based on octave-band SPLs (Reference 1).

The computation of source octave-band SPLs included the effect of spherical spreading but not that of atmospheric attenuation, since the latter is a frequency dependent quantity. The contribution of each source was summed, giving the octave-band spectrum for the whole vehicle. At this point the effect of standard day atmospheric attenuation was included, the PNL being calculated from the resulting octave band SPLs.

For the on-ground condition, determination was made of the maximum PNL on a 500 foot circle centered at the aircraft. The operating condition was maximum power just prior to brake release (STOLs) or to lift-off (VTOLs).

The PNL for the fly-over condition was determined at a point beneath the flight path 5000 feet from brake release (or lift-off). The aircraft were operating at take-off power which defined the flight profiles shown in Figure 3-3. The flight paths used for the VTOL aircraft were take-off without a vertical climb segment, typical of airport operation. These flight paths are shown in Figure 3-3 which also shows the flight paths with 400-foot vertical climb segments.

The 400-foot climb segment would have a small effect on DOC (about 2% for a 500 mile stage length), small increase in fuel and gross weight, and some reduction in noise as shown in Figure 3-7.

The noise sources present on the various V/STOL aircraft were analyzed as follows:

1. Propeller and Rotor Rotational Noise:
The SPL of the fundamental and higher harmonics of rotational noise were obtained by adjusting measured data. The adjustments were based on Gutin calculations of the SPL of the fundamental or first harmonic of blade passage (rotational) noise; one calculation was for the conditions of the measured data, the other for the conditions of the vehicle being studied. The difference between the measured and calculated SPL of the fundamental gave the discrepancy to be expected from the theory. This correction term was applied to the calculated SPLs of the various configurations, where applicable. The SPLs of the harmonics were obtained from the dB difference between the harmonics and the fundamental of the measured data. These dB differences for the harmonics were applied to the adjusted fundamental calculated for each configuration. The Gutin equation, in engineering terms, (Equation 1 of Reference 2) is shown below:

$$p = \frac{169.3mB\ M_t}{2sA}\left[\frac{550\ P_H}{c(0.8M_t)^2} - T\cos\beta\right] J_{mB}(X)$$

Where:

A = disc area = $\pi D^2/4$ (ft^2)

B = number of blades

c = velocity of sound (ft/sec)

D = diameter (ft)

$J_{mB}(X)$ = Bessell function of the first kind of order mB and argument X

m = harmonic number = 1,2,3,....

M_t = tip Mach number

p = r.m.s. pressure (dynes/sq.cm.)

P_H = horsepower

s = field point distance (ft)

T = Thrust (lb.)

X = Argument of Bessel function = $0.8\ M_t mB \sin\beta$

β = angle to field point measured from the direction of thrust

The measured data used to correct the calculated SPLs were adapted as follows:

(a) Propellers: The flight data in Reference 3 were evaluated for trends in SPL at different power settings. The first three octave bands appeared to be dominated by the first three harmonics of propeller rotational noise. These results are presented in Figure 3-4a along with the calculated SPL of the fundamental. Figure 3-4b presents similar data for the measured on-ground SPLs and the corresponding calculated SPL for the fundamental.

(b) Rotors: The spectrum analysis of Figure 15, Reference 4, was used to obtain the SPLs for the main and tail rotors. The fundamental for the main rotor is not shown, but a level was obtained by extrapolation from the second and third harmonics. The SPLs for the main rotor are presented in Figure 3-5a, the tail rotor in Figure 3-5b.

2. Vortex Noise:

The overall sound pressure level (OASPL) of the vortex noise was calculated directly (Equation 2, Reference 3). The equation is derived from the work of Yudin:

$$\overline{I}_v = 10 \log_{10} \left[\frac{K A_B\ V_{0.7}^6}{10^{-16}} \right]$$

FIGURE 3-4

MEASURED PROPELLER SOUND PRESSURE LEVELS

(a) Flight Data (for one propeller)

Harmonic Number	SPL (dB)
m = 1	76.5
m = 2	79
m = 3	75

Calculated SPL for m = 1: 86 dB

Conditions:

$B = 4$

$D = 13.5$ ft

$P_H = 2600$ HP/propeller

$s = 1000$ ft

$V_t = 720$ ft/sec

(b) On-Ground Data (for one propeller)

Harmonic Number	SPL (dB)
m = 1	104
m = 2	101
m = 3	100.5

Calculated SPL for m = 1: 103 dB

Conditions (different from those in (a) above):

$P_H = 3360$ HP/propeller

$s = 170$ ft

FIGURE 3-5

MEASURED HELICOPTER ROTOR ROTATIONAL NOISE

(a) Main Rotor

Harmonic Number	SPL (dB)
m = 1	(94)
m = 2	92
m = 3	90
m = 4	86
m = 5	81
m = 6	80

Calculated SPL for m = 1: 78 dB

(b) Tail Rotor

Harmonic Number	SPL (dB)
m = 1	83
m = 2	84
m = 3	82
m = 4	79
m = 5	76
m = 6	73

Calculated for m = 1: 73 dB

(c) Conditions for both main rotor and tail rotor data

$B = 2$

$D = 43.75$ ft./8.4 ft

$P_H = 450$ HP/50 HP

$s = 200$ ft.

$V_t = 720$ ft/sec/710 ft/sec.

NOTE: Double numbers are for main rotor/tail rotor, respectively.

Where:

A_B = total blade area

$\overline{I}_v$ = overall rms sound pressure level at 300 feet (dB)

K = 3.8×10^{-27} (emperical constant)

$V_{0.7}$ = blade section speed at 0.7 radius (ft/sec)

The equation for the frequency (f_{max}) at which the vortex noise spectrum peaks (Figure 7, Reference 5) is:

$$f_{max} = S \frac{V_H}{L_{0.7}}$$

Where:

S = Strouhal number

$L_{0.7}$ = effective airfoil thickness at the 0.7 radius station (ft)

V_H = helical tip speed (ft/sec)

The Strouhal number used in the above reference (0.126) is for the "near field". Von Gierke (Reference 6) states that 0.185 has been determined experimentally to be the Strouhal number for typical propellers. This number has been used in the calculations for the noise sensitivity analysis.

3. Jet Noise:

The OASPL for jet noise was calculated directly by the methods in Reference 6. The OASPL for one engine at a 200 foot sideline distance is obtained from the equation:

$$\text{OASPL} = 10 \log f(V_R) + 10 \log \rho^2 A$$

Where:

$f(V_R)$ is given in Figure 1, Reference 6

ρ = density of gas = $\frac{(W)}{(A)\ (V_J)}$

V_J = jet velocity (ft/sec)

W = weight flow (lb/sec)

V_A = aircraft velocity (ft/sec)

$V_R = V_J - V_A$ = relative velocity

A = nozzle area (ft^2)

OASPL = overall sound pressure level (dB)

Figure 2, Reference 7 presents two spectra for jet noise from circular nozzles, one for on-ground conditions, the second for flight conditions. For rectangular slot nozzles, such as those used for the jet flap, the results of Reference 8 indicate that the OASPL is the same as would be predicted for a circular nozzle of the same total area; however, the spectrum appears to be defined by an effective nozzle diameter of twice the slot height. This modification was incorporated in the frequency calculation for all rectangular and circular (annulus) slot nozzles.

4. Turboshaft Engine Exhaust Noise:

In processing the flight data of Reference 3, the fourth through seventh octave band SPLs appear to be power dependent. The data for the eighth octave band indicate the possibility of the presence of a discrete frequency which is not power dependent. Since this may not be typical of the engines which would be used in the V/STOL configurations, the SPL of the eighth octave band was obtained by extrapolation of the SPLs of the sixth and seventh octave bands instead. The resulting SPLs are presented in Figure 3-6a. The on-ground data were taken from measurement points behind and to the side of the engines since exhaust noise predominates there. Only idle and full power conditions, were presented. It was assumed that the noise would be power dependent, as in the case of the flight data. The data for full power are presented in Figure 3-6b.

FIGURE 3-6

MEASURED TURBOSHAFT ENGINE EXHAUST NOISE

(a) Flight Data (for one engine)

Octave Band	SPL (dB)
300 - 600	73
600 - 1200	69.5
1200 - 2400	68
2400 - 4800	66.5
4800 - 9600	65

Conditions: 2600 HP/engine, 1000 ft. altitude

Note: The effects of atmospheric attenuation have been removed from the measured data.

(b) On-Ground Data (for one engine)

Octave Band	SPL (dB)
300 - 600	91.5
600 - 1200	86.5
1200 - 2400	89.5
2400 - 4800	88.5
4800 - 9600	84.5

Conditions: 3360 HP/engine, 170 ft. radial distance

5. Lift Fan Noise:

The lift fan noise calculation method (Reference 9) uses an emperical relationship involving energy flux. The calculation yields the PWL of the fan blade passage noise. The SPL was obtained from the LWL assuming spherical spreading. The equations used are:

$$A_a = \frac{\pi}{4} (D_T)^2 \left[1 - \frac{(D_H)^2}{D_T} \right]$$

$$T_T = T + \Delta T$$

$$E = \frac{(H_T)\ (W)}{A_a}$$

Figure 13, Reference 8, is used to obtain the quantity PWL - 10 log $\frac{(A_a)\ (n)}{(N_r)} \left[\frac{D_H}{D_T} \right]^2$ for the above calculated E.

Solving the above equation gives the PWL from which the OASPL is calculated. The terms are defined as:

A_a = active fan area (ft^2)

T = inlet temperature (^{o}R)

ΔT = stage temperature rise (^{o}R)

T_T = exit temperature (^{o}R)

H_T = total enthalpy (Btu/lb) from gas tables

W = weight flow (lb/sec)

D_H = inner diameter (ft)

D_T = outer diameter (ft)

n = R.P.M.

N_r = number of fan blades

The SPLs of the harmonics were taken from Figure 15, Reference 9 which plots the SPL of the harmonics relative to the OASPL.

The results of the noise sensitivity study are summarized in Figure 3-7. A discussion of the contributing noise sources and the results of the analysis for the individual aircraft follows:

1. Deflected Slipstream:

The major noise sources are the propeller rotational noise and the turboshaft engine exhaust noise. The results indicate that reduction in tip speed is offset by increases in power required so that negligible reductions occur in the PNL. The results are presented in Figures 3-8, 3-9, and 3-10. The aircraft decisions are discussed in LR 19585, Appendix C.

2. Jet Flap:

The high-velocity, small area multiple jet nozzles are the primary noise sources. As would be expected, the lower power of the 2000 foot STOL results in lower on-ground noise; however, the higher fly-over altitude of the 1000 foot STOL results in a lower fly-over noise at the 5000 foot location. Figures 3-11, 3-12, and 3-13 present the results for the jet flap.

3. Fan-In-Wing:

The major noise source at close distances is the fan blade passage. The SPL and frequency from this source is essentially the same for both aircraft; as a result the on-ground PNL is the same for both aircraft. However, this high frequency fan noise will be subjected to rapid attenuation with increasing distance, due to atmospheric absorption. This accounts for the sizeable differences in PNL for the fly-over at the 5000 foot point (SPL differences are approximately 7dB due to spherical spreading and 4dB due to atmospheric absorption). These effects are greater as altitude differences increase. The results for the fan-in-wing are presented in Figures 3-14, 3-15, and 3-16.

4. Tilt Rotor:

The major noise sources are the rotor rotational noise and the turboshaft engine exhaust noise. Reductions in tip speed appear to be beneficial. The increased noise from increases in power are more than offset by the reduction in noise as tip speed is reduced, resulting in a net noise reduction. Figures 3-17, 3-18 and 3-19 present the results for the tilt rotor.

5. Stopped Rotor Prop:

The main rotor and tail rotor rotational noise and the turboshaft engine noise constitute the major noise sources for the stopped-rotor prop configurations examined. At both the on-ground and fly-over locations the propellers were not providing forward thrust and thus they did not contribute to the noise calculated at these locations. The trend of PNL with decreasing tip speed is unexpected since the 800 fps version has a higher PNL than either the 900 or 700 fps configurations. Reducing the rotor tip speed increases both the power required and the noise output but reduces the blade passage frequency. For the 700 fps version the reduction in blade passage frequency moved one of the more intense harmonics of rotational noise outside the human audible range. Consequently the PNL for this version is lower than that for the 800 and 900 fps configurations. The results for the stopped rotor prop configurations are presented in Figures 3-20, 3-21, and 3-22.

FIGURE 3-7

RESULTS OF THE NOISE SENSITIVITY ANALYSIS

AIRCRAFT TYPE	MODEL	GROSS WEIGHT (lb)	D.O.C. (cents/seat mile)	BLOCK SPEED (knots)	CRUISE SPEED (knots)	PERCEIVED NOISE LEVEL (PNdB) AT BRAKE RELEASE	PERCEIVED NOISE LEVEL (PNdB) 5000 ft FROM B.R.	ALTITUDE 5000 ft FROM B.R. (feet)
Deflected Slipstream	900 fps	46,900	1.96	244	283	100	96	610
	800 fps	48,000	1.90	259	311	99	95	610
	700 fps	49,300	1.857	271	335	99	95	610
Jet Flap	1000-ft STOL	77,700	2.9	353	478	130	115	1335
	2000-ft STOL	63,200	2.3	369	483	128	123	570
Fan-In-Wing	VTOL	DOES NOT APPLY TO THIS STUDY						
	1000-ft STOL	67,900	2.67	383	493	105	99	690
	2000-ft STOL	59,000	2.475	364	475	105	111	322
Tilt Rotor	900 fps	65,000	2.67	296	363	111	96/98*	2025/1625*
	800 fps	67,900	2.70	296	363	109	93/96*	2020/1620*
	700 fps	76,400	3.09	291	365	108	90/93*	2000/1600*
Stopped Rotor Prop	900 fps	71,000	2.65	312	400	109	92/95*	2080/1680*
	800 fps	75,000	2.70	315	410	110	93/96*	2090/1690*
	700 fps	85,700	3.12	322	437	109	91/94*	2190/1790*

* Double numbers refer to altitudes for the two different takeoff profiles (see text). The lower PNL refers to the higher altitude.

NOTE: All perceived noise levels rounded to nearest whole PNdB.

FIGURE 3-8

DEFLECTED SLIPSTREAM
SENSITIVITY OF CHARACTERISTICS TO PROPELLER TIP SPEED

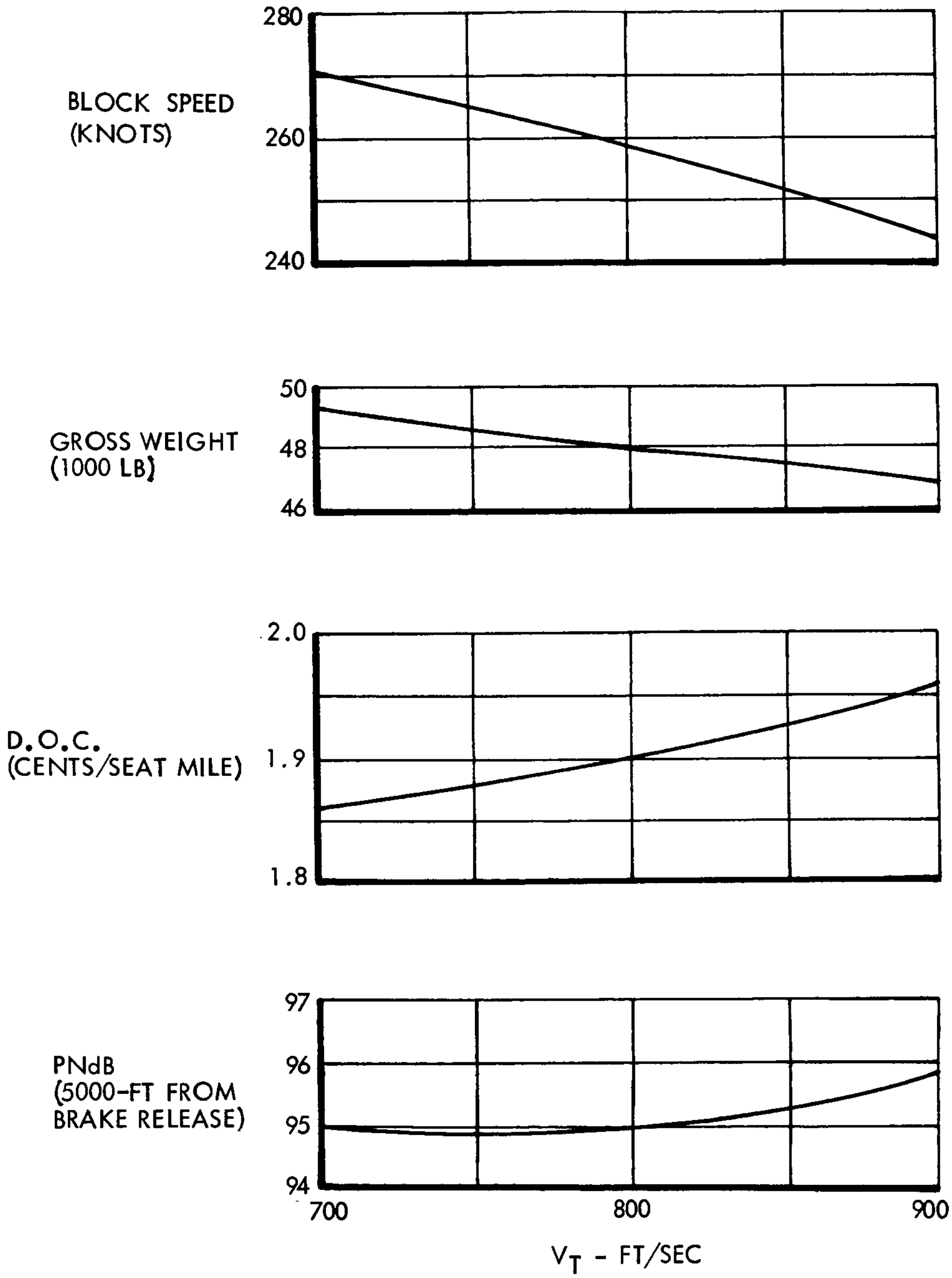


FIGURE 3-9

DEFLECTED SLIPSTREAM

SENSITIVITY OF CHARACTERISTICS TO NOISE REDUCTION

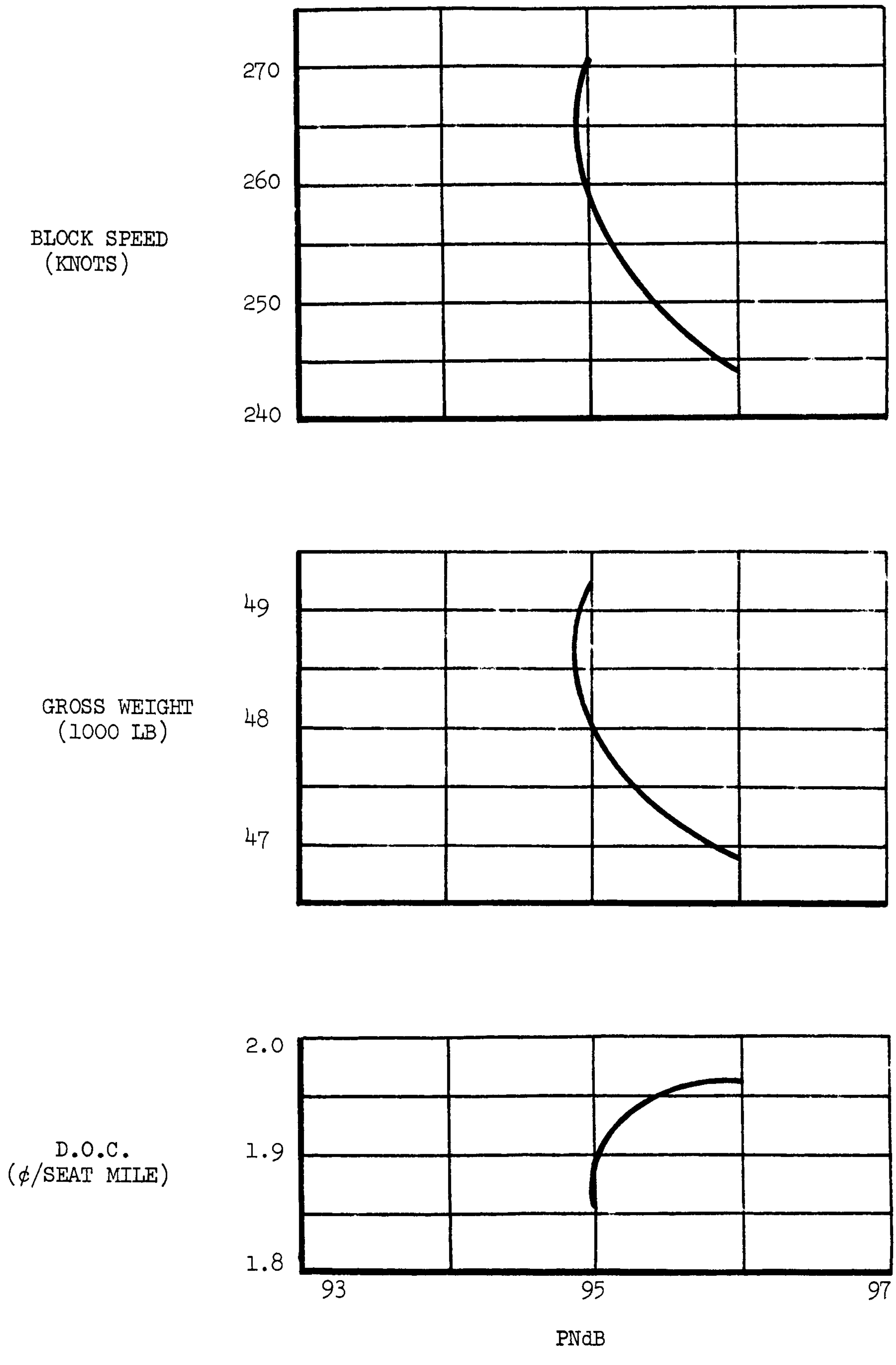


FIGURE 3-10

DEFLECTED SLIPSTREAM

SENSITIVITY OF CHARACTERISTICS TO TIP SPEED

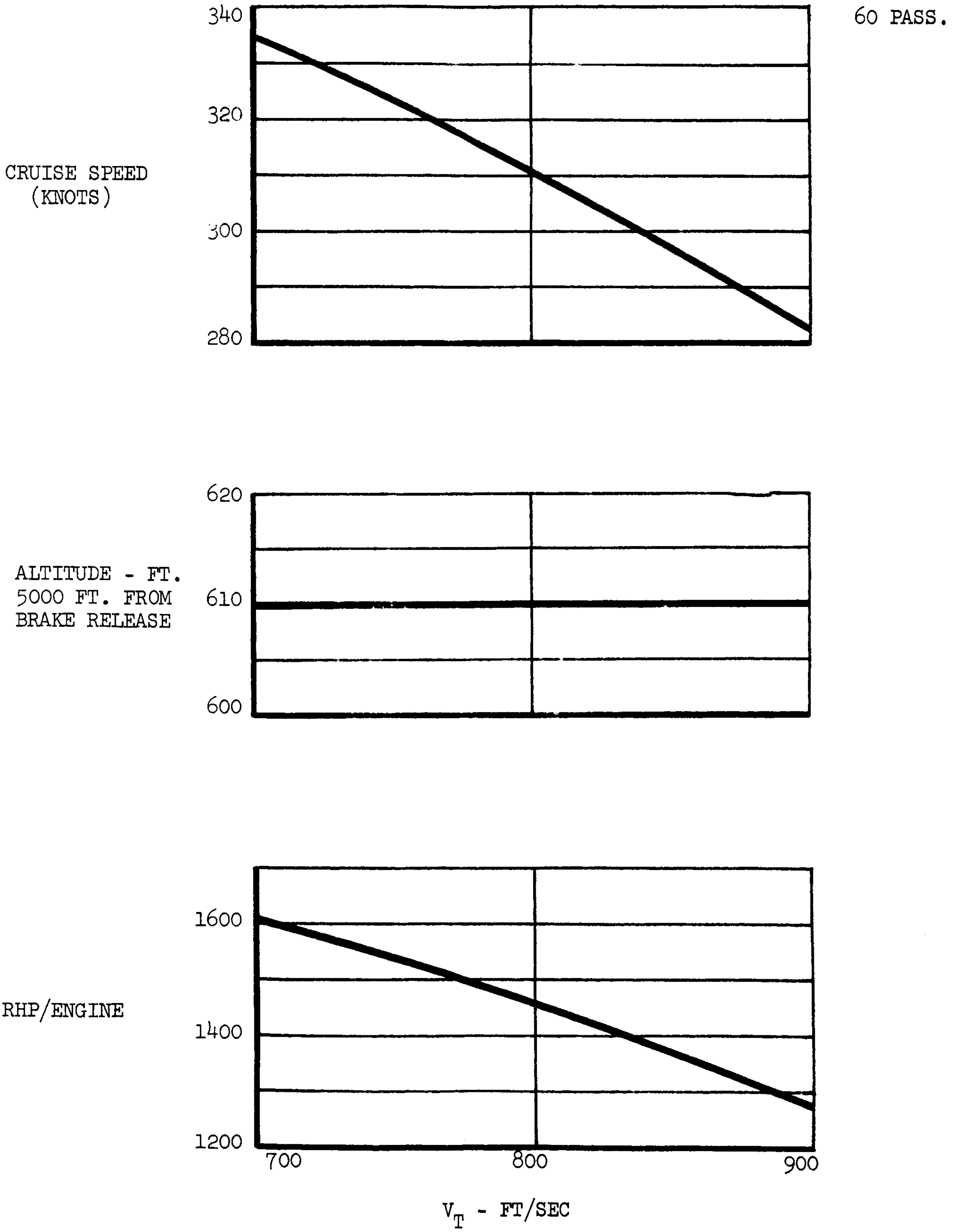


FIGURE 3-11

JET FLAP

SENSITIVITY OF CHARACTERISTICS TO T/W_{STATIC}

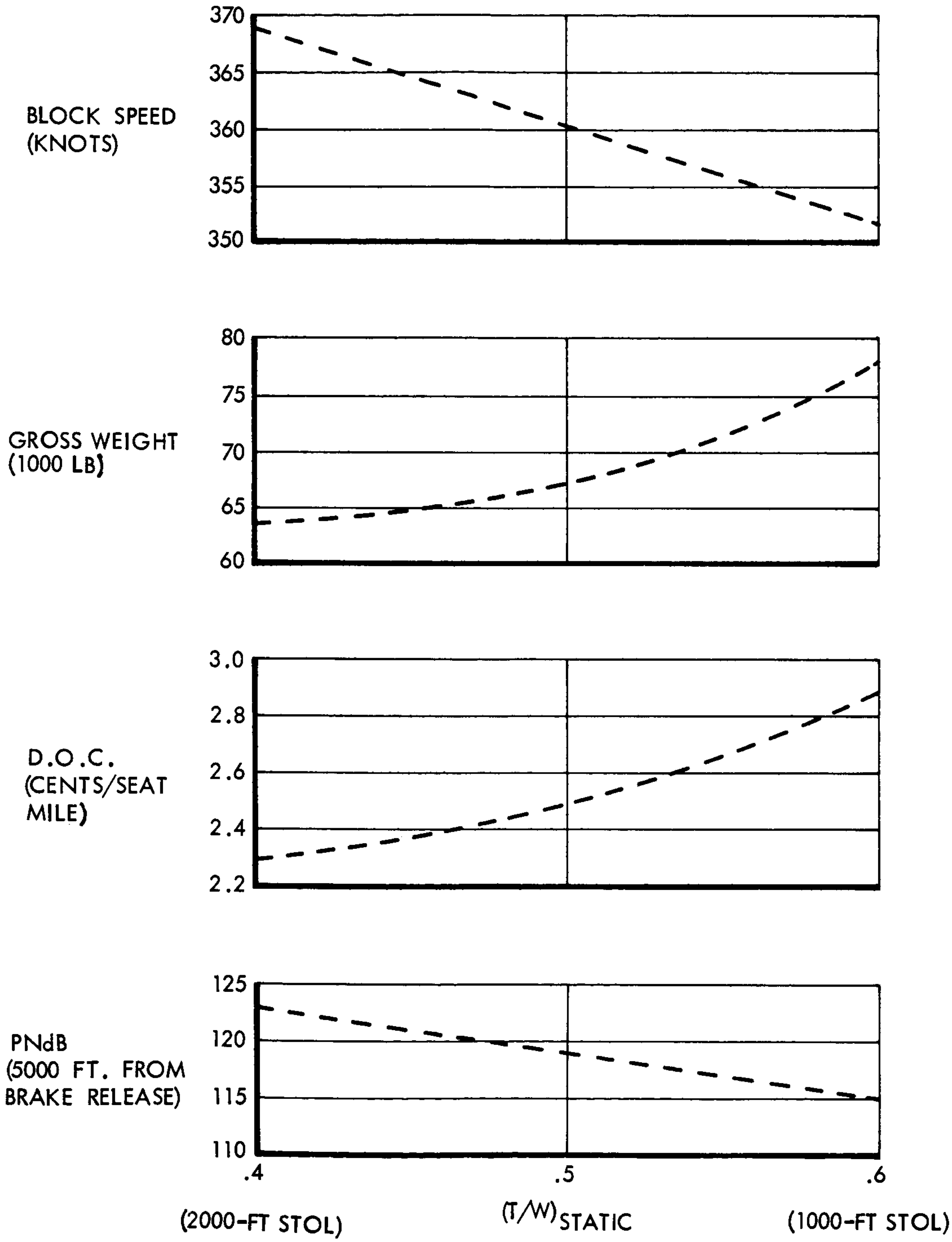


FIGURE 3-12

JET FLAP

SENSITIVITY OF CHARACTERISTICS TO NOISE REDUCTION

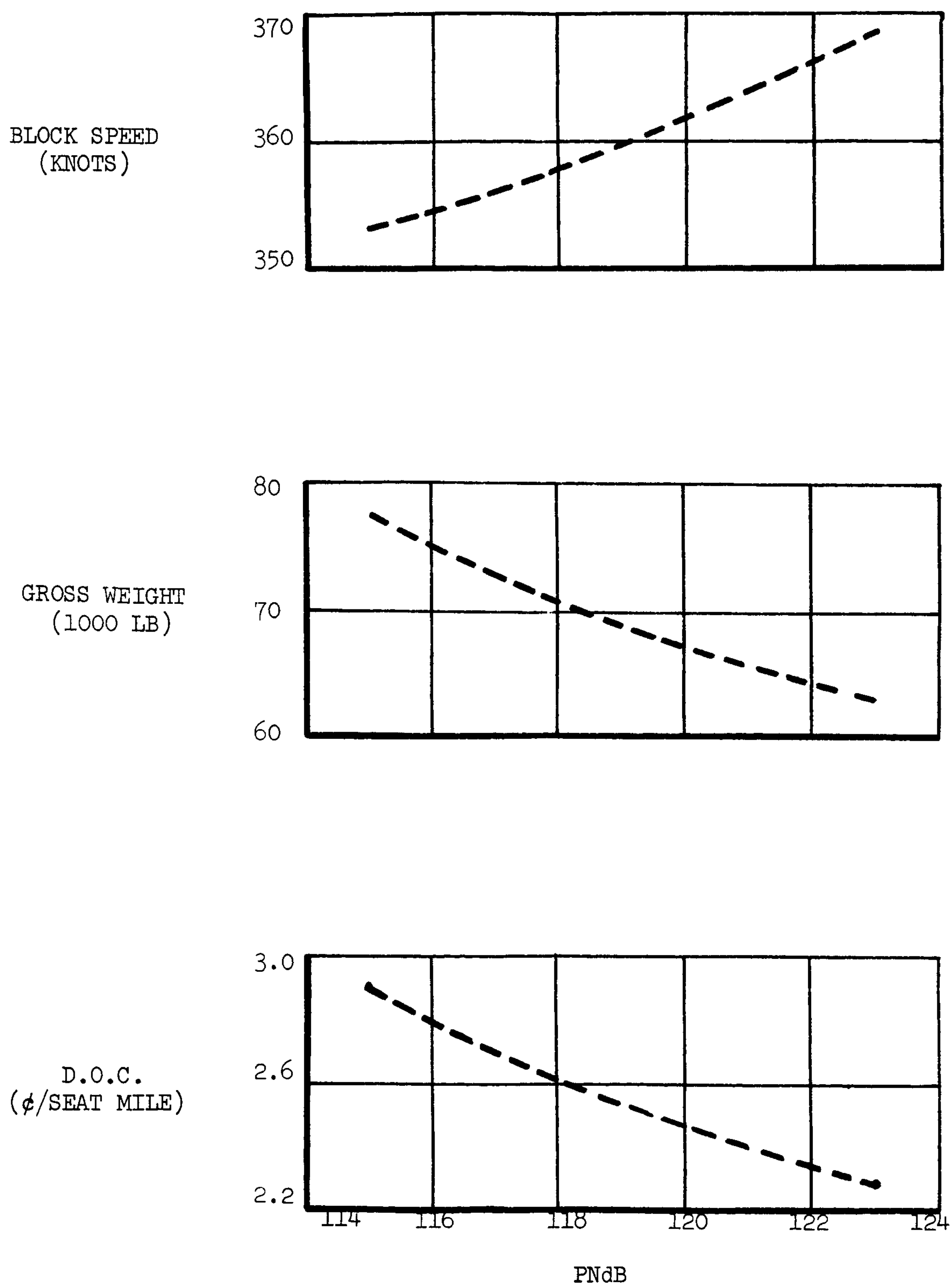


FIGURE 3-13

JET FLAP

SENSITIVITY OF CHARACTERISTICS TO $(T/W)_{STATIC}$

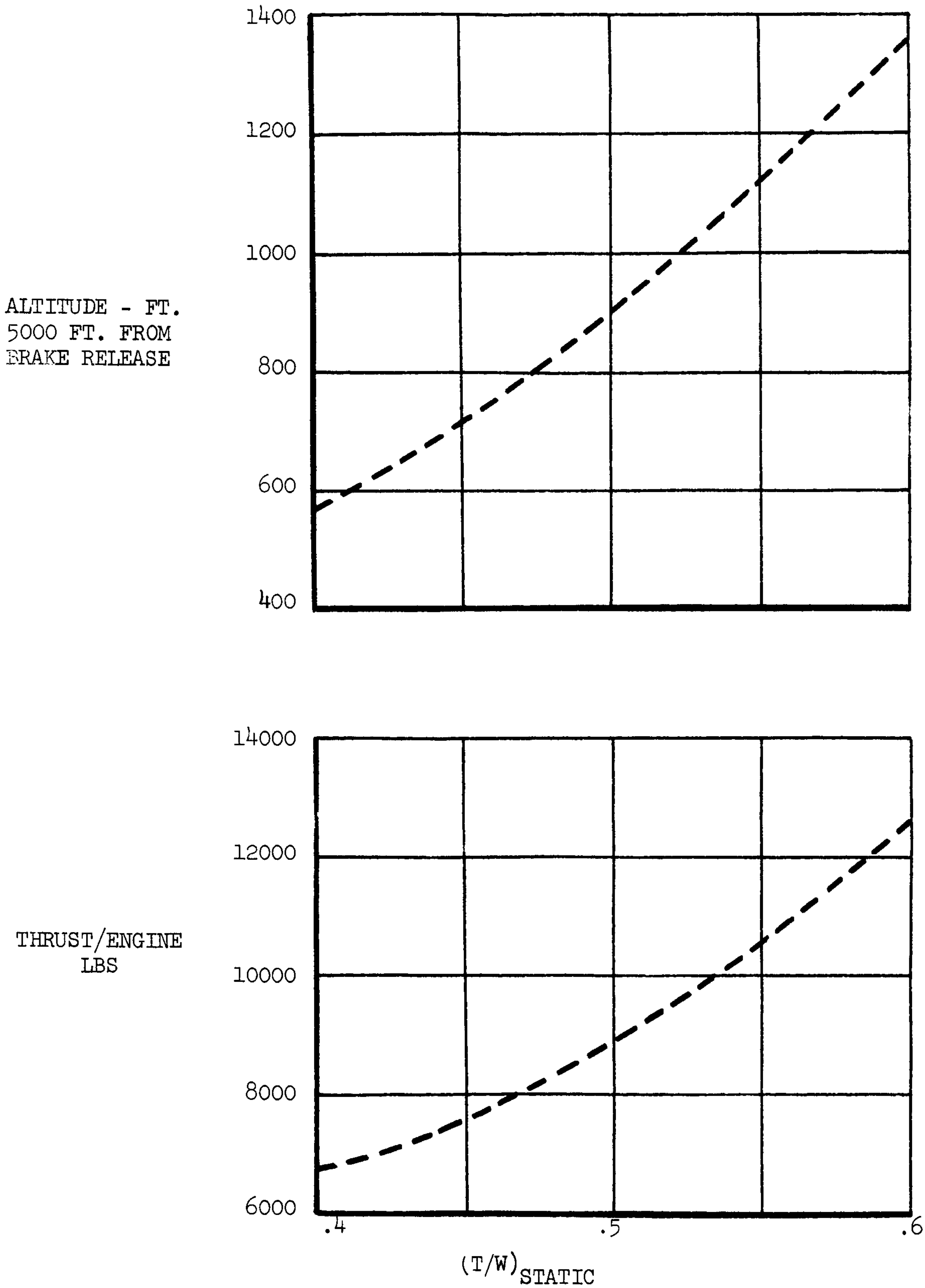


FIGURE 3-14

FAN-IN-WING
SENSITIVITY OF CHARACTERISTICS TO $(T/W)_{STATIC}$

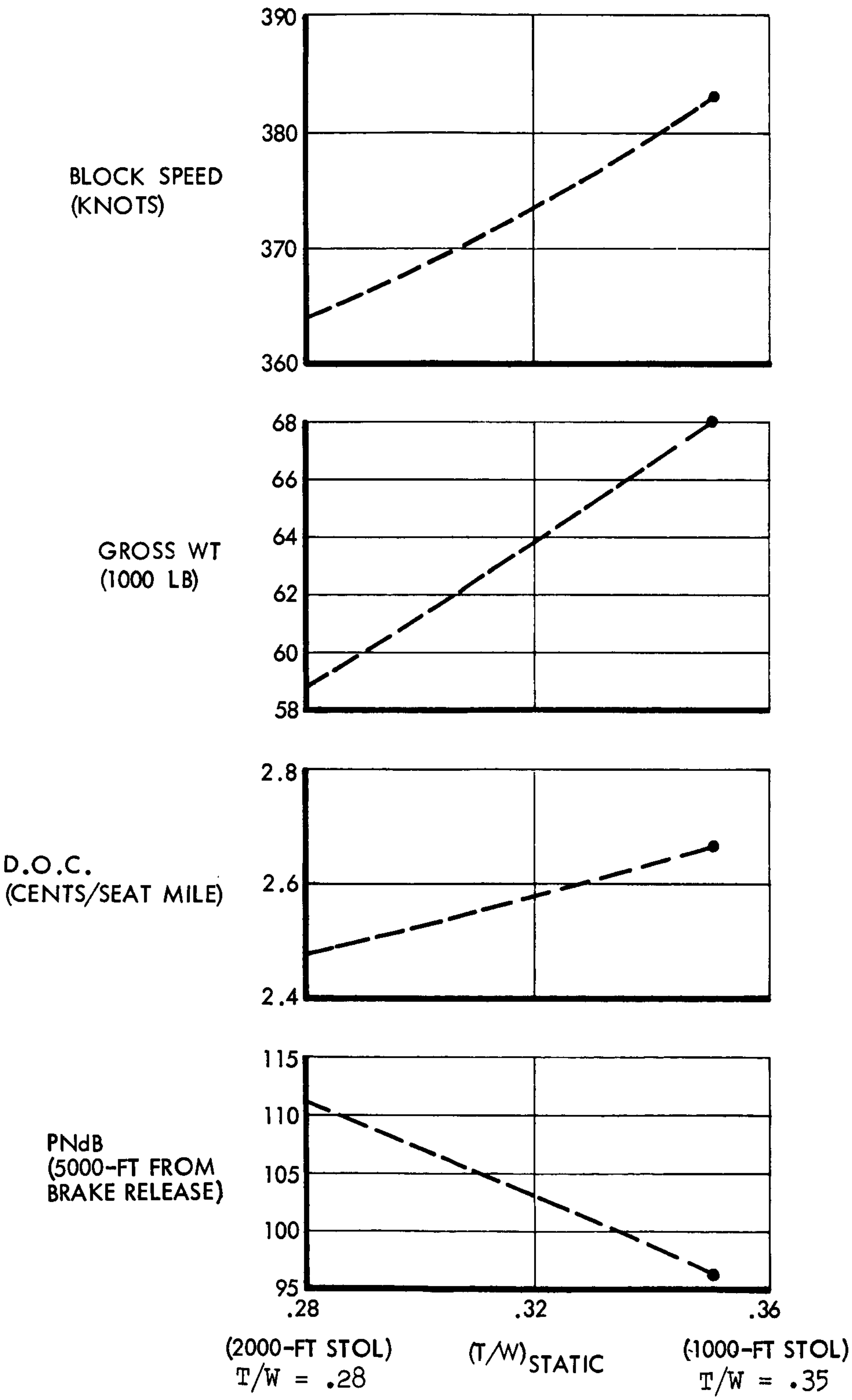


FIGURE 3-15

FAN-IN-WING

SENSITIVITY OF CHARACTERISTICS TO NOISE REDUCTION

60 PASS.

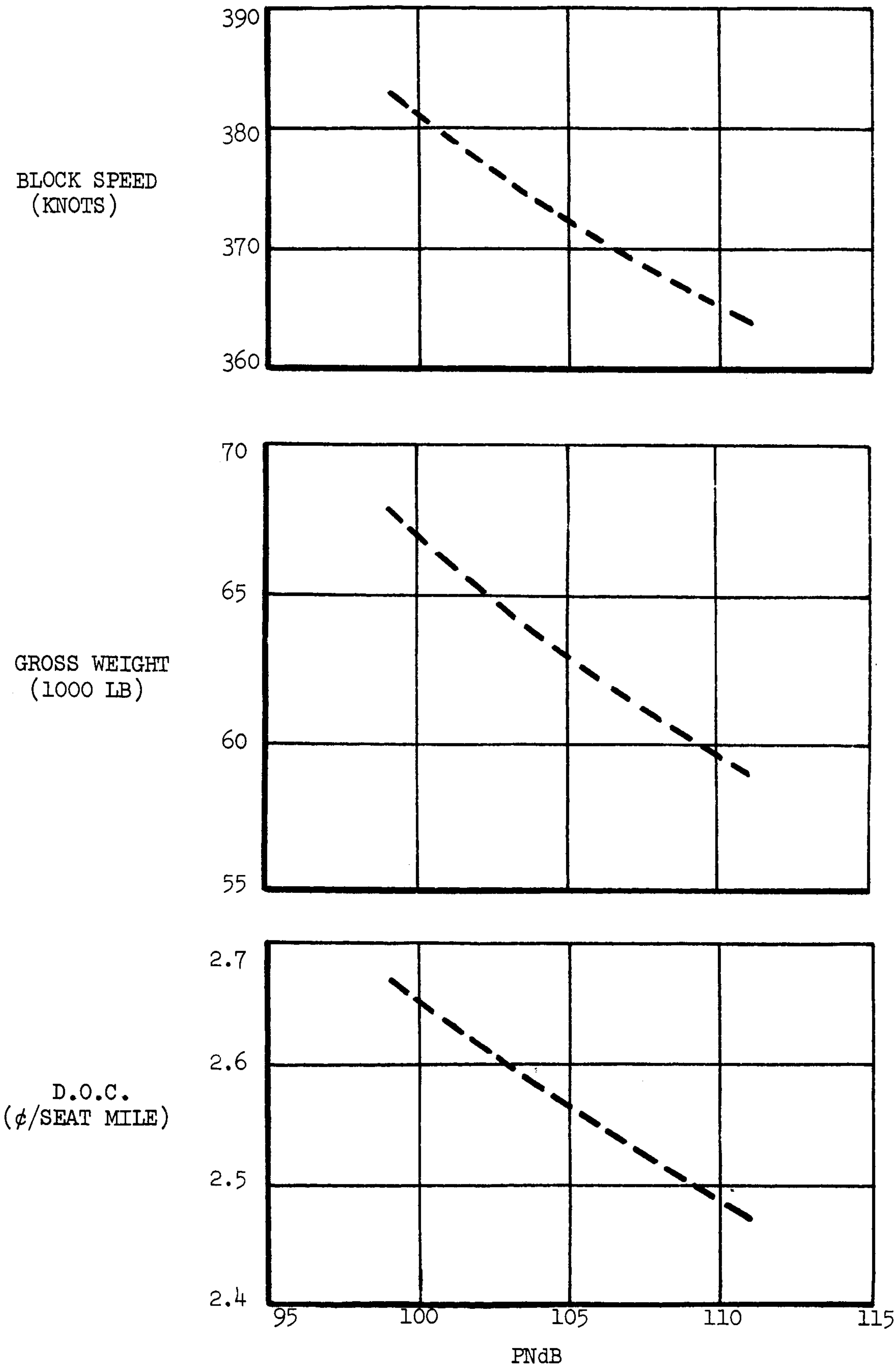


FIGURE 3-16

FAN-IN-WING

SENSITIVITY OF CHARACTERISTICS TO $(T/W)_{STATIC}$

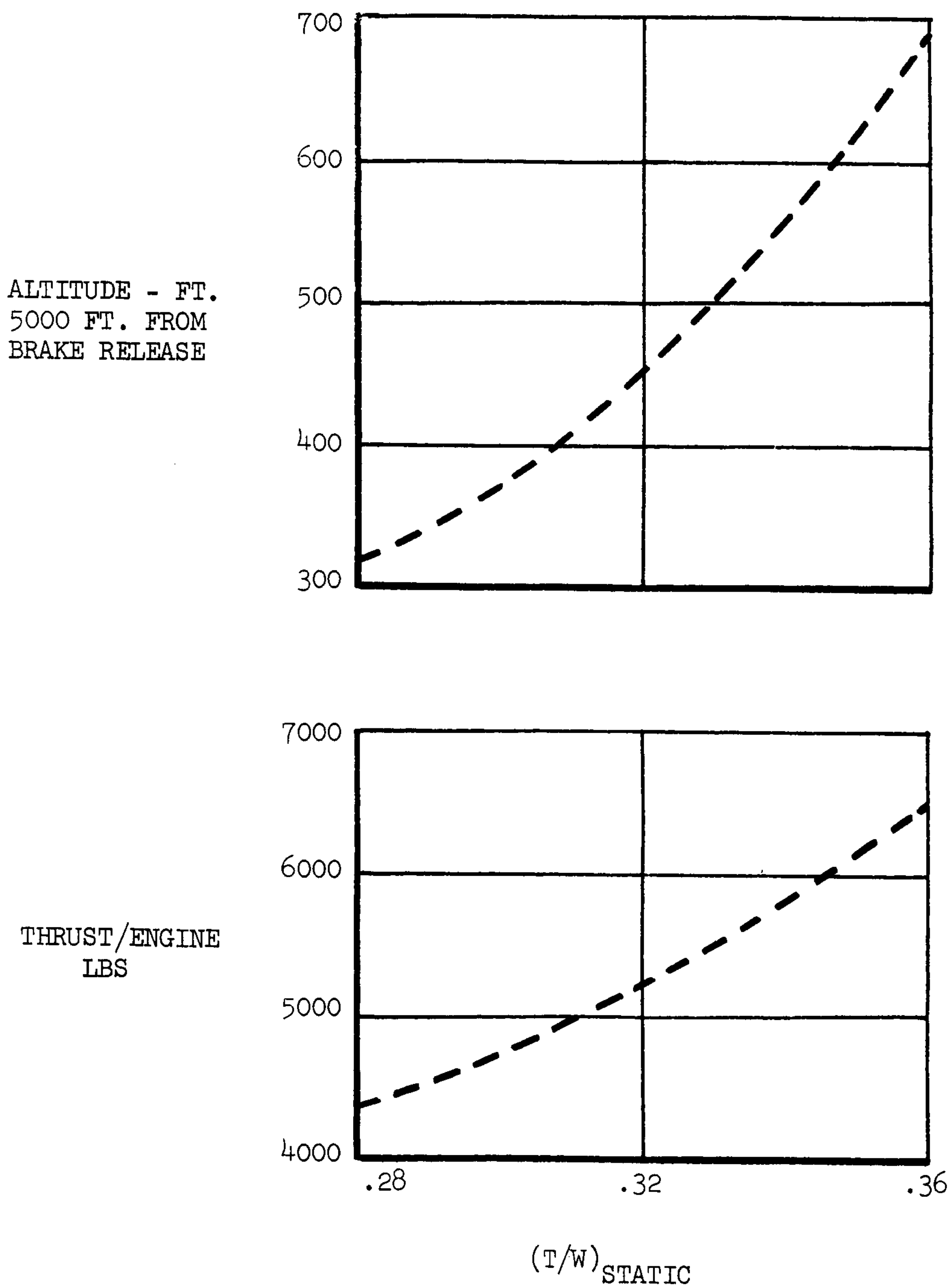


FIGURE 3-17

TILT ROTOR
SENSITIVITY OF CHARACTERISTICS TO ROTOR TIP SPEED

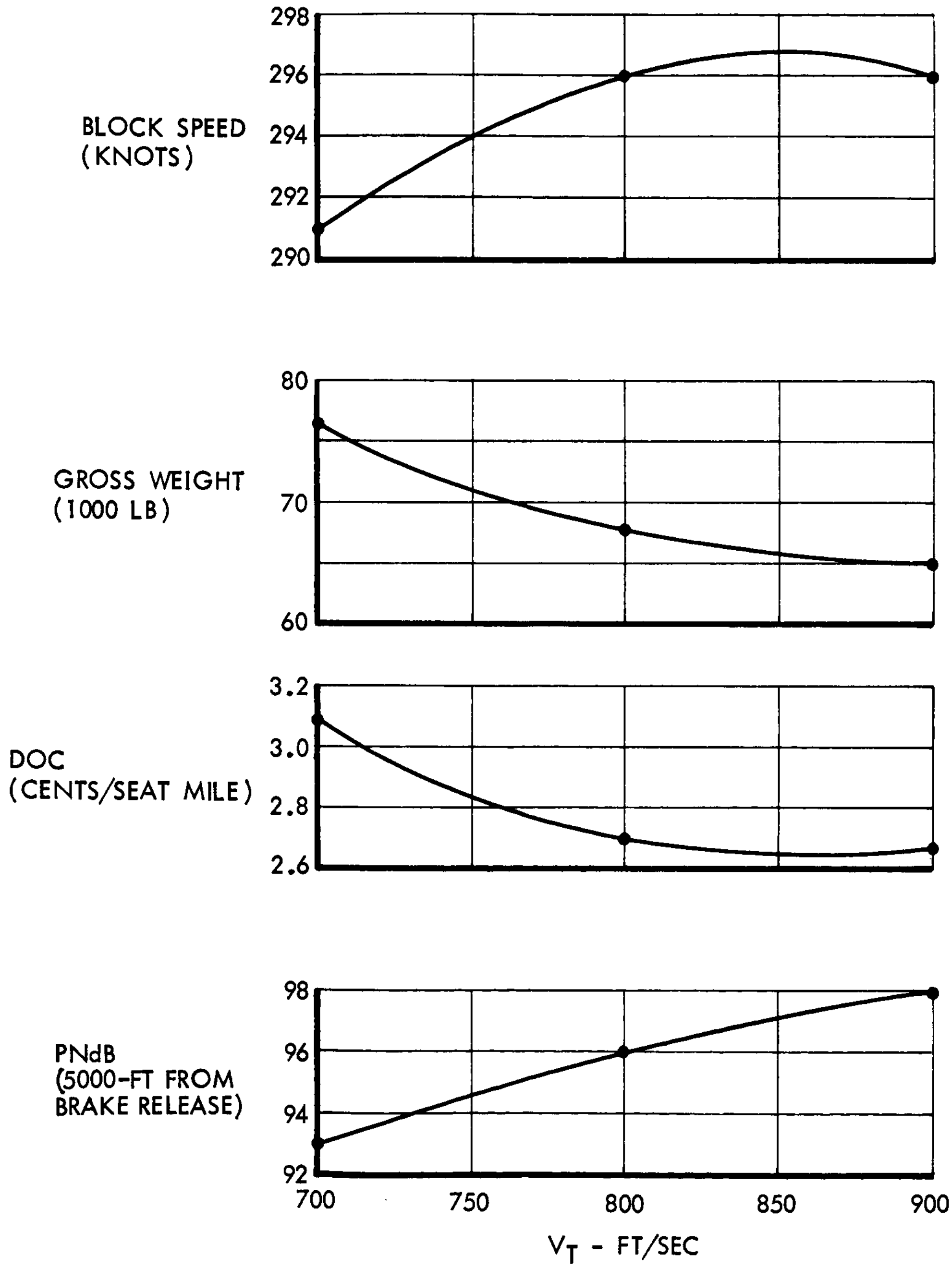


FIGURE 3-18

TILT ROTOR

SENSITIVITY OF CHARACTERISTICS TO NOISE REDUCTION

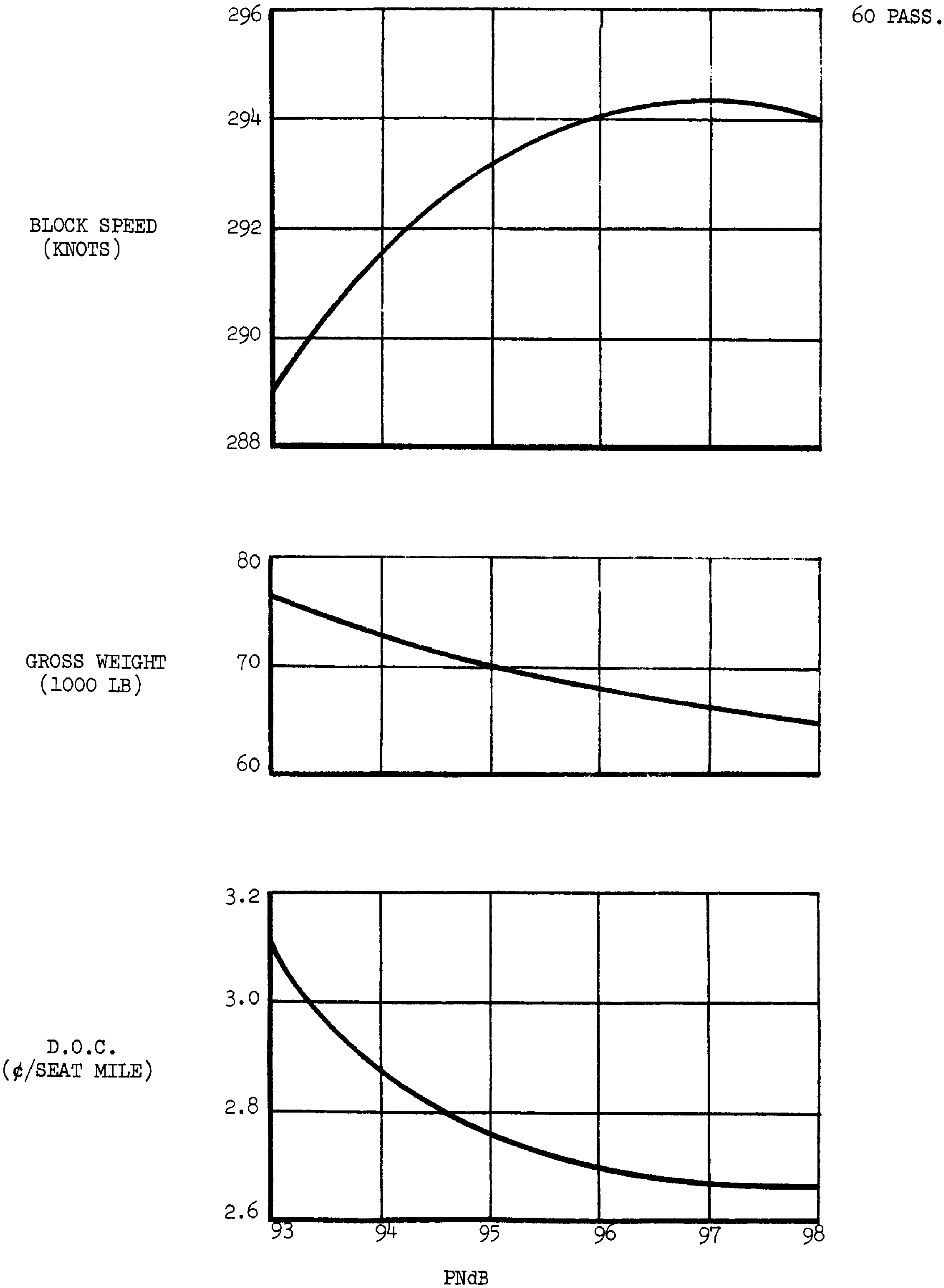


FIGURE 3-19

TILT ROTOR

SENSITIVITY OF CHARACTERISTICS TO TIP SPEED

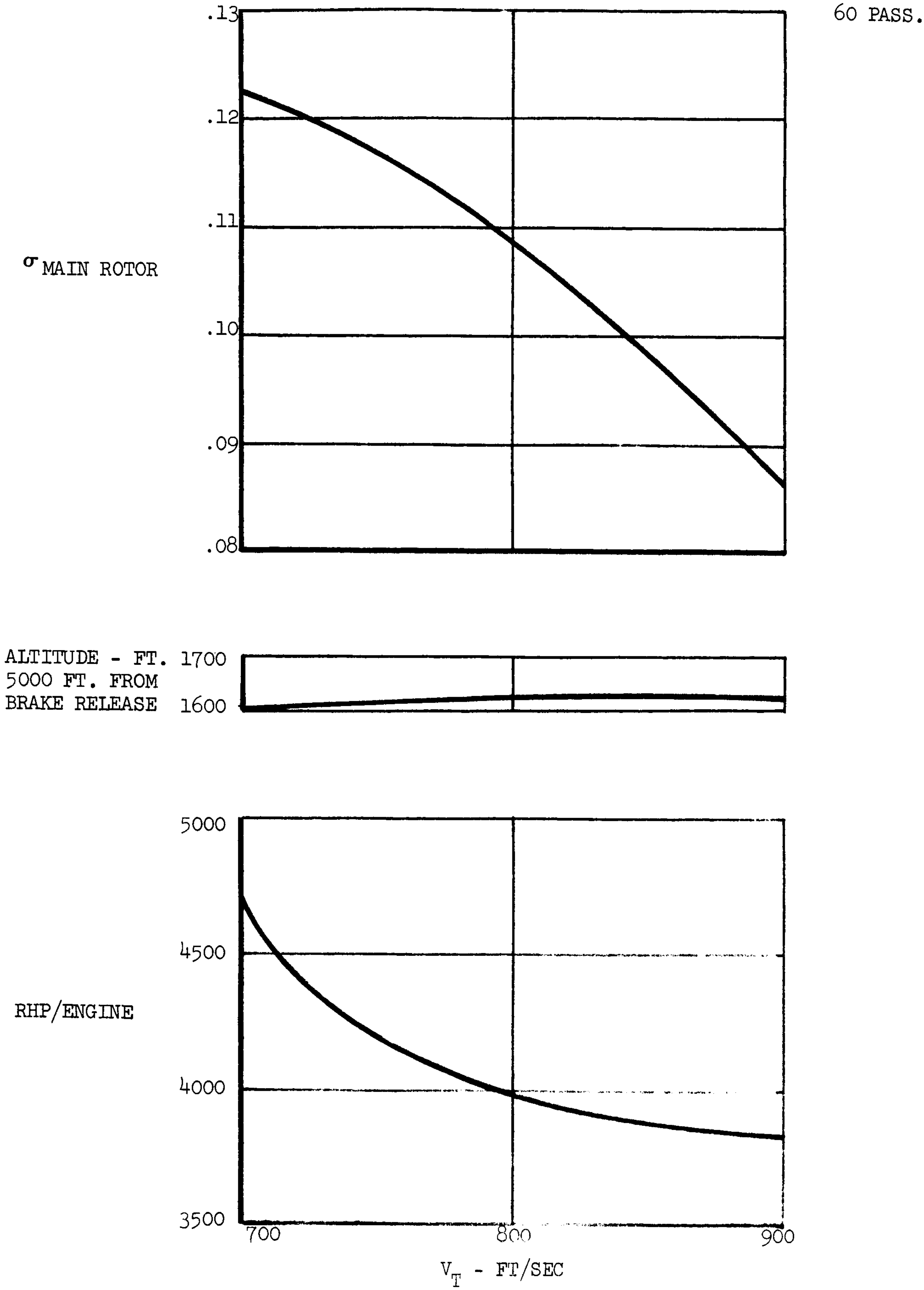


FIGURE 3-20

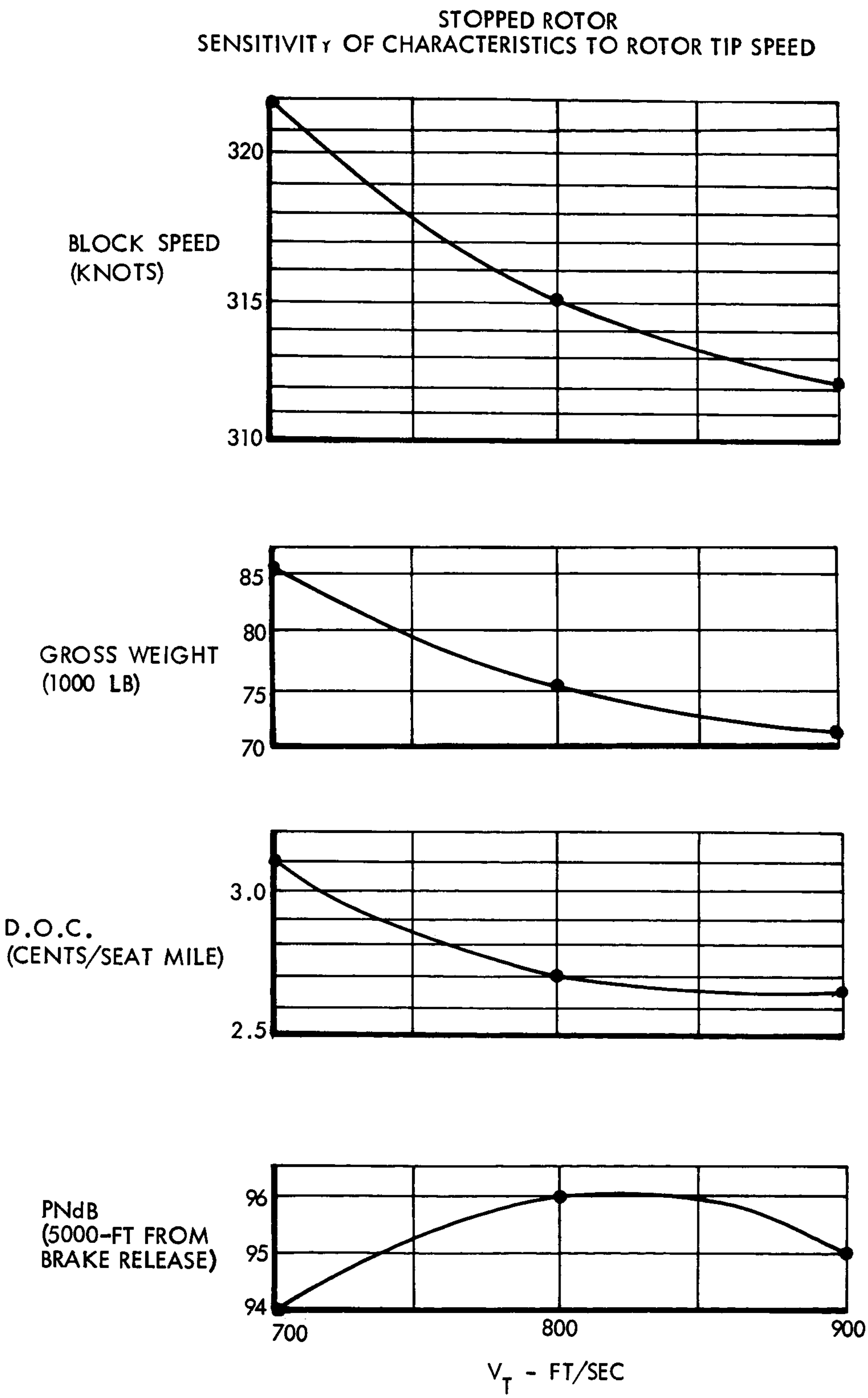

STOPPED ROTOR
SENSITIVITY OF CHARACTERISTICS TO ROTOR TIP SPEED
BLOCK SPEED (KNOTS)
320
315
310
GROSS WEIGHT (1000 LB)
85
80
75
70
D.O.C. (CENTS/SEAT MILE)
3.0
2.5
PNdB (5000-FT FROM BRAKE RELEASE)
96
95
94
700
800
900
V_T - FT/SEC

FIGURE 3-21

SINGLE STOWED ROTOR

PROPELLER DRIVEN

SENSITIVITY OF CHARACTERISTICS TO NOISE REDUCTION

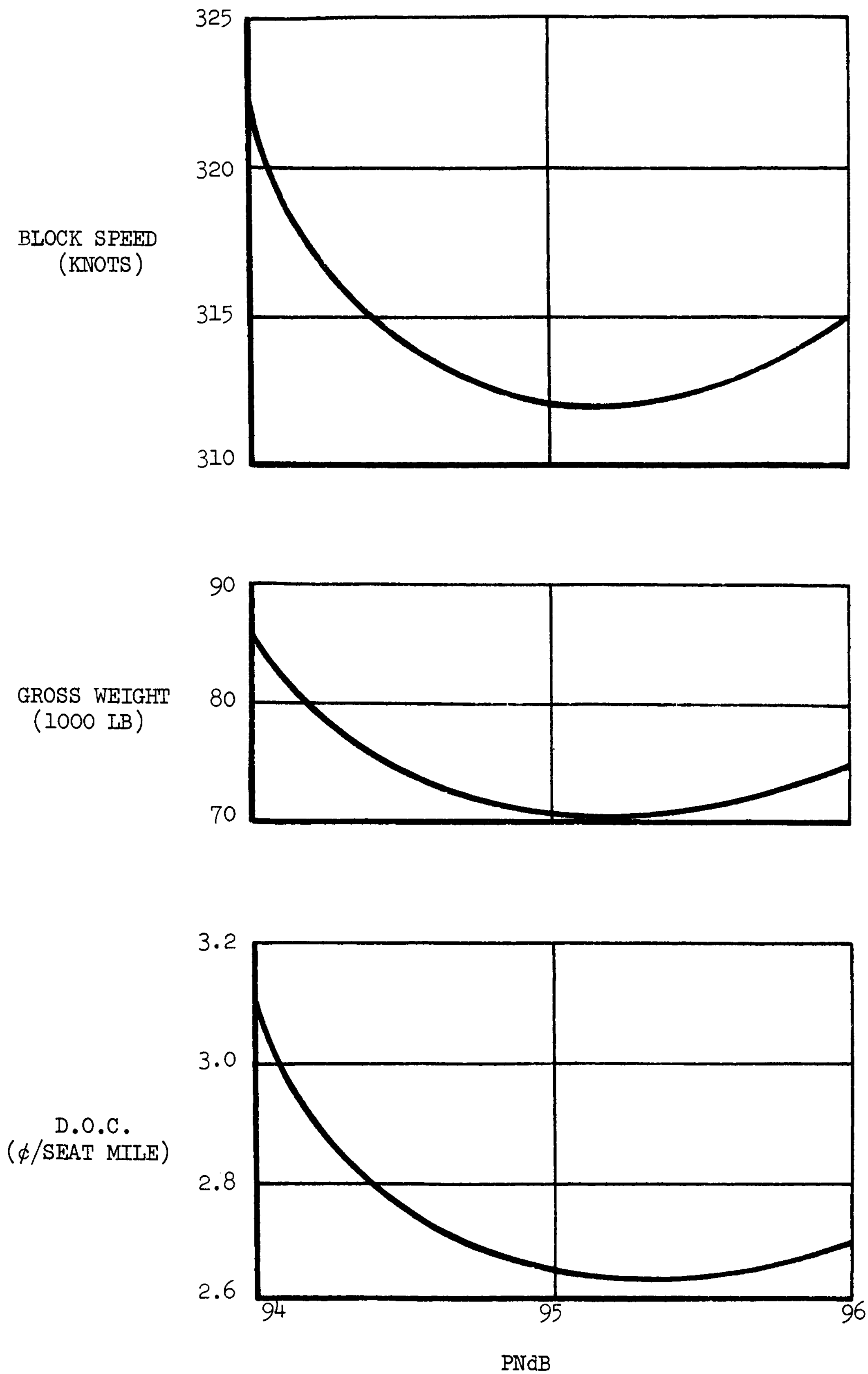
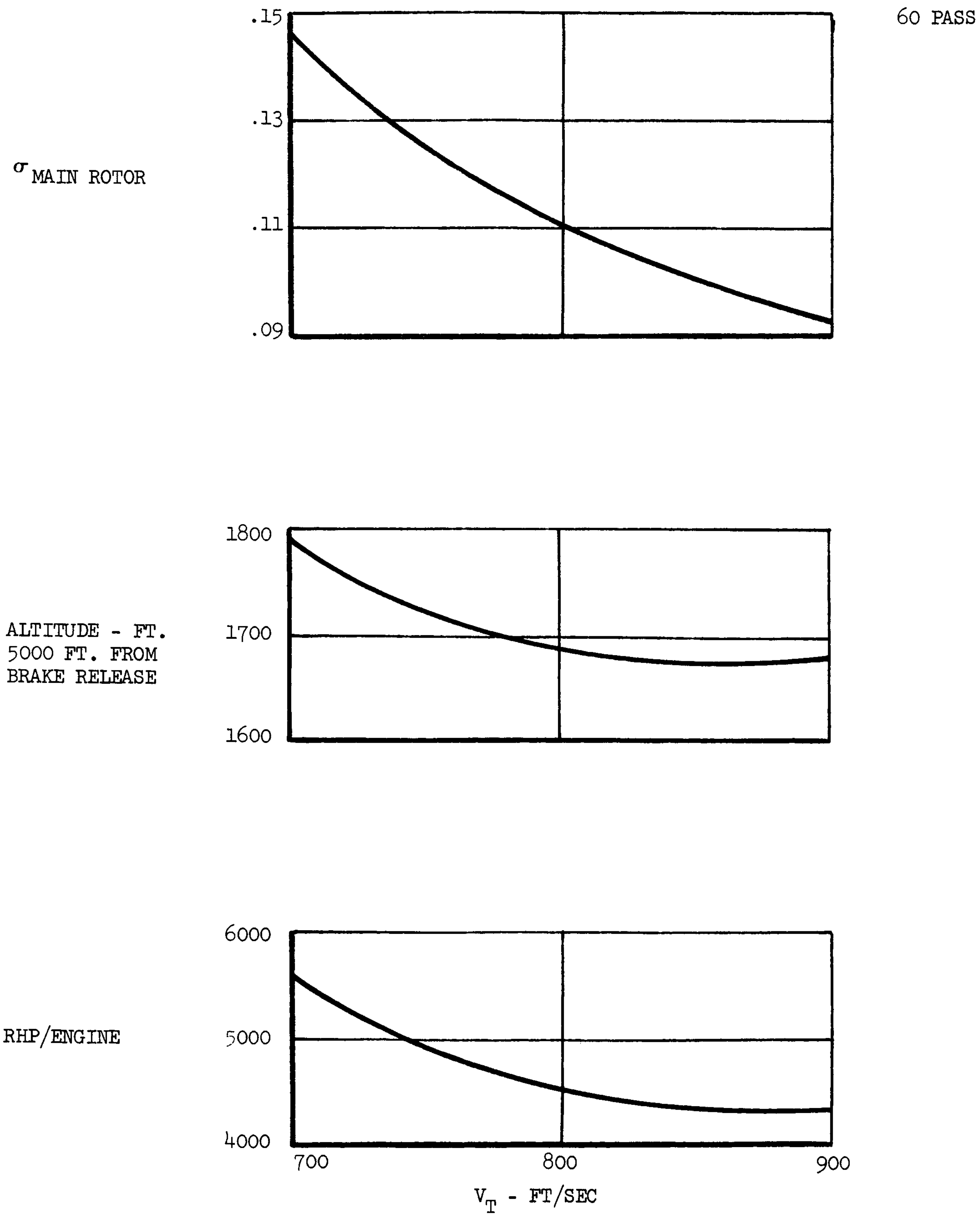


FIGURE 3-22

STOPPED ROTOR PROP

SENSITIVITY OF CHARACTERISTICS TO TIP SPEED



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